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INVENTORY

Development Test I Advanced Attack Helicopter Competitive Evaluation Hughes YAH-64 Helicopter Final Rpt. Dec. '76

DOCUMENT IDENTIFICATION

USAAEFA Project No. 74-07-2

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USAAEFA PROJECT NO. 74-07-2

DEVELOPMENT TEST 1

ADVANCED ATTACK HELICOPTER COMPETITIVE EVALUATION

HUGHES YAH-64 HELICOPTER

FINAL REPORT

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Field	Entry
1. REPORT NUMBER	USAAEFA PROJECT NO. 74-07-2
2. GOVT ACCESSION NO.	
3. RECIPIENT'S CATALOG NUMBER	
4. TITLE (and Subtitle)	DEVELOPMENT TEST I ADVANCED ATTACK HELICOPTER COMPETITIVE EVALUATION HUGHES YAH-64 HELICOPTER
5. TYPE OF REPORT & PERIOD COVERED	FINAL REPORT 9 July - 30 September 1976
6. PERFORMING ORG. REPORT NUMBER	USAAEFA PROJECT NO. 74-07-2
7. AUTHOR(s)	MAJ ROBERT L. STEWART, MAJ RICHARD C. TARR, LT MICHAEL HILL, BARCLAY H. BOIRUN, TOM P. BENSON, EDWARD E. BAILES, WILLIAM Y. ABBOTT
8. CONTRACT OR GRANT NUMBER(s)	
9. PERFORMING ORGANIZATION NAME AND ADDRESS	US ARMY AVIATION ENGINEERING FLIGHT ACTIVITY EDWARDS AIR FORCE BASE, CALIFORNIA 93523
10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS	46-6-R0079-01-46-EC
11. CONTROLLING OFFICE NAME AND ADDRESS	US ARMY AVIATION ENGINEERING FLIGHT ACTIVITY EDWARDS AIR FORCE BASE, CALIFORNIA 93523
12. REPORT DATE	DECEMBER 1976
13. NUMBER OF PAGES	383
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)	
15. SECURITY CLASS. (of this report)	UNCLASSIFIED
15a. DECLASSIFICATION/DOWNGRADING SCHEDULE	N/A
16. DISTRIBUTION STATEMENT (of this Report)	**DISTRIBUTION STATEMENT A** **Approved for public release; Distribution Unlimited**
17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)	
18. SUPPLEMENTARY NOTES	
19. KEY WORDS (Continue on reverse side if necessary and identify by block number)	Advanced attack helicopter Development Test I Hughes YAH-64 helicopter Performance and handling qualities evaluation

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The United States Army Aviation Engineering Flight Activity conducted a performance and handling qualities evaluation of the Hughes Helicopter Company YAH-64 as part of the advanced attack helicopter Development Test I. Flight tests were conducted from 9 July through 30 September 1976 at Edwards Air Force Base, California (elevation 2303 feet), Bishop, California (elevation 4120 feet), and Coyote Flats, California (elevation 9500 feet). Two instrumented test aircraft were

(continued)

20. Abstract

used: Army serial number 74-22248 and Army serial number 74-22249. A total of 87 flights for 92.2 hours (62.0 productive) were flown on the two aircraft. The performance evaluation included hover, vertical climbs, forward flight climbs, level flight, autorotational descent, lateral acceleration, and vertical displacement. The YAH-64 failed to meet three of the performance requirements of the Army systems specification: the cruise airspeed requirement by 4 knots true airspeed (KTAS), the single-engine service ceiling requirement by 350 feet, and the vertical rate of climb requirement by 266 feet per minute. The mission gross weight was determined to be 14,242 pounds, 1042 pounds in excess of the Hughes Helicopter Company specification weight. Sixteen enhancing characteristics were identified, the most significant of which were the handling qualities with the automatic stabilization equipment disengaged; the low levels of physiologically significant vibrations at the crew stations; automatic engine torque matching; the tail wheel configuration; autorotational descent performance and rotor speed control; the automatic radio transmitter selecter system; and the digital fiberoptic instruments. Seven deficiencies were identified: the loss of directional control in 25 to 30 KTAS left sideward flight; tail rotor horsepower exceeding the limit value when hovering in a right crosswind; the structurally inadequate tail rotor drive shaft-mounted cooling fan; recurring starting problems of both the auxiliary power unit and the main engines; the lack of preset frequencies on the UHF and VHF radios; and the Marconi fuel quantity gauge zero shift. Sixty-four shortcomings were identified, the most significant of which were the restricted field of view due to canopy structure and distortion; the blast shield precluded passing items from pilot to copilot; the inadequate wheel brake effectiveness and difficulty of operation; the pitch-up tendency between hover and 60 KCAS; and restricted forward field of view due to nose-high attitude. Although a number of cockpit shortcomings were identified, the YAH-64 cockpit design was a significant improvement over existing attack helicopters. Numerous envelope limits were imposed during this evaluation which would be unacceptable for an operational aircraft. The YAH-64 has the potential to be developed into an excellent attack helicopter. Although the scope of this test did not include a complete check of all aircraft specifications for compliance, eight additional items of noncompliance with the Army systems specification requirements were identified.

DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
P O BOX 209, ST. LOUIS, MO 63166

DRDAV-EQ

24 MAR 1978

SUBJECT: Directorate for Development and Engineering Position on the Final Report of USAAEFA Project No. 74-07-2, Government Competitive Test Advanced Attack Helicopter (AAH), Hughes YAH-64 Helicopter, December 1976

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1. The subject report represents an evaluation of the YAH-64 in its early stage of development. As a result, numerous flight envelope limitations (defined in the Safety-of-Flight Release) were imposed which would be unacceptable for an operational attack helicopter. These limitations precluded evaluation of the aircraft in many significant areas. Also, because of the overweight status of the prototype aircraft and many other performance improvements being incorporated into the Phase 2 design, the performance levels shown in the subject report are not representative of the AH-64 operational capability. The incorporation of the TADS/PNVS and HELLFIRE missile systems in lieu of the simulated TOW missile system utilized in the Government Competitive Test will likely result in significant performance variations from the prototype aircraft. This letter represents the Directorate for Development and Engineering's position on the subject report. Its purpose is to point out:

a. The fact that significant flight envelope limitations were required should be considered a deficiency.

b. Those areas where improvements have been incorporated in the Phase 2 design to eliminate deficiencies/shortcomings identified by the test results of the prototype aircraft.

c. Those areas where we differ in the significance of the test results.

The material is proceeded by reference to the appropriate paragraph of the conclusions and recommendations followed by specific comments or other miscellaneous comments.

2. Paragraph 147b: Significant flight envelope limitations precluding evaluation of the aircraft in many areas are as follows:

a. Rapid acceleration to the left was prohibited due to possible loss of directional control.

b. Total engine power torque limitations significantly below intermediate rated power for speeds above 80 KCAS in level flight with further reductions for maneuvering flight, i.e., only 14% of total torque was allowed when maneuvering above 1.5g at high speeds.

c. Right sideward flight speed limitations at heavy gross weight ranging from 5 to 15 knots depending on altitude.

d. Never exceed level flight speed limitations limiting the aircraft to less than the maximum speed at level flight at intermediate rated power.

Design changes are incorporated in the Phase 2 configuration, which if effective, will eliminate the need for these envelope limitations.

3. Paragraph 147c: The high control system loads mentioned here should be significantly reduced by the incorporation of the swept tip main rotor blade in the Phase 2 configuration. In addition, the strength of the flight control system actuator is being increased.

4. Paragraph 147d: The Phase 2 aircraft incorporates larger effective tail rotor diameter and other vertical tail changes which are intended to alleviate abnormal cross-wind limitations.

5. Paragraph 147f: The inclusion of the radar altimeter is planned for the AH-64 production configuration.

6. Paragraph 147h: The single-point pressure refueling and defueling system and the automatic T4.5 limiting feature of the YT700-GE-700 engine were shown as major enhancing features in the UTTAS evaluation report conclusions. It is felt that the presence of the same features in an attack helicopter are equally enhancing and should have been reflected in paragraph 147h of the YAH-64 report.

7. Paragraph 147h(1): The excellence of the SAS off handling qualities is grossly overstated. For example, these characteristics were of sufficient concern that three specific CAUTION notes were added to the Safety-of-Flight Release based on evaluation of the contractor's flying

qualification data during the development phase. The AEFA evaluation did not cover the high speed end of the flight envelope. Subsequent flight tests in the high speed regime have shown that the Phase 1 configured aircraft exhibits unsatisfactory flight characteristics. At the airspeeds evaluated, it is apparent that the HQRS ratings shown tend to reflect the superior skill of a test pilot(s) rather than the performance that can be expected from a standard operational pilot.

8. Paragraph 147h(2): The 4/rev vibration increase at airspeed below 70 KCAS is of major concern. Planned tactical deployment of the HELLFIRE missile will be during low speed operation approaching hovering flight. Some of the vertical vibration data at these low speeds greatly exceeds the .05g maximum allowable vibration level per the requirements document.

9. Paragraph 148a: The larger tail rotor and other vertical tail changes being incorporated in the Phase 2 design previously discussed are intended to alleviate this deficiency.

10. Paragraph 148b: Tail rotor drive system design ratings are being increased and appropriate design changes are being made so that the Phase 2 configuration should not limit the low speed agility of the aircraft.

11. Paragraph 148c: With the incorporation of the Black Hole IR suppressor concept, the tail rotor drive shaft mounted cooling fan has been eliminated from the Phase 2 configuration.

12. Paragraphs 148d, 148e, 148g: These problems were eliminated by Phase 2 design changes.

13. Paragraph 148f: As with other Army air items, some avionics will not have preset frequencies since this is not considered a requirement by DA. However, this position is being reconsidered due to the significant nap-of-the-earth operation of the attack helicopters.

14. Paragraph 149: Some of the shortcomings listed are not consistent with the AAH requirements documentation; others have been accepted as deviations to the requirements by the AAH Source Selection Evaluation Board (SSEB). Comments are offered below on several AEFA listed shortcomings which are considered to be more serious or for which corrective action has not been taken.

DRDAV-EQ 24 [illegible]

SUBJECT: Directorate for Development and Engineering Position on the Final Report of USAAEFA Project No. 74-07-2, Government Competitive Test Advanced Attack Helicopter (AAH), Hughes YAH-64 Helicopter, December 1976

a. Paragraph 149e: The braking system of the AAH has not been changed for Phase 2 since the system meets MIL Spec requirements. However, redesign to the crew station will make brake application easier and should alleviate the problems mentioned in paragraphs 149f and 149ff.

b. Paragraph 149h: This paragraph states that five minutes to recharge the hydraulic accumulator is a shortcoming. This time is well within the Government System Specification requirements of two starts within a ten minute period and as such should not be considered a shortcoming. Problems experienced during the AEFA tests which required any hand pumping are considered abnormal. A review of the problems indicated several quality control and independent failures related to the specific APU and not related to the overall design. Phase 2 design has a totally different APU system.

c. Paragraph 149j: Improvement in the forward flight field-of-view over the nose during climbs is not considered practical and may be slightly worsened when the TADS/PNVS is incorporated.

d. Paragraph 149v: The larger VSD utilized results in the increased pilot scan as mentioned. However, a small VSD would also represent an undesirable cockpit configuration. We, therefore, believe the basic flight instrument arrangement in optimum and the slight increase in scan time acceptable.

e. Paragraph 149y: In addition, the static longitudinal instability causing extensive pilot workload during maneuvering flight as indicated by an HQRS of 6 is also considered a shortcoming.

f. Paragraph 149kk: Contractor testing prior to GCT verified the same large longitudinal trim requirements resulting from power changes which admittedly caused extensive pilot workload. These were evaluated as HQRS of 6 by their pilots and should be considered a deficiency.

g. Paragraph 149yy: The heat generated from the subject CAUTION lights is not considered a meaningful shortcoming. Military Specifications allow a maximum surface temperature of 105°F with 25% of the bulbs in the CAUTION panel illuminated. Laboratory tests indicated for the 28 volt system of 100% of the bulbs illuminated the AAH surface temperatures were 112°F. It is apparent that a 75% reduction in illuminated bulbs would reduce the measured 112°F temperature to below the specification requirement. In addition, the human touch pain threshold is considered to be greater than 115° to 117°F. Consideration is being given to incorporation of a 5 volt lighting system in the production design.

24 MA[illegible] [illegible]3

DRDAV-EQ
SUBJECT: Directorate for Development and Engineering Position on the Final Report of USAAEFA Project No. 74-07-2, Government Competitive Test Advanced Attack Helicopter (AAH), Hughes YAH-64 Helicopter, December 1976

15. Appropriate action has been taken on all recommendations contained in paragraphs 151-154.

16. As stated in paragraph 1, the AEFA evaluation was conducted on a flight envelope restricted prototype aircraft and was not representative of the Phase 2 design. The AAH SSEB negotiated corrections to be incorporated in the Phase 2 design to all significant problems encountered during the Government Competitive Test. For a complete definition of the Phase 2 AAH configuration and its performance capabilities refer to AMC-SS-AAH-H10000 dated 23 November 1976, which is available from the Advanced Attack Helicopter Project Manager's Office, PO Box 209, St. Louis, Missouri 63166.

FOR THE COMMANDER:

Walter A. Ratcliff

WALTER A. RATCLIFF
Colonel, GS
Director of Development
and Engineering

PREFACE

The engineering flight test portion of the advanced attack helicopter Phase I development test was conducted by the United States Army Aviation Engineering Flight Activity (USAAEFA) at Edwards Air Force Base and at remote test sites at China Lake Naval Air Facility, Bishop, and Coyote Flats, California. The extensive coordination necessitated by the use of the remote test sites would have been extremely difficult without the helpful and effective assistance of numerous USAAEFA and China Lake support personnel, and Edwards AFB firemen and medics. Throughout the testing phase in Bishop, the utmost cooperation and assistance was rendered to the test team by the Bishop Airport manager and his support personnel.

The test aircraft was maintained and the test instrumentation was supplied, calibrated, and maintained by Hughes Helicopter Company personnel. A special thanks to the technical representatives from the Sperry Rand Corporation and the General Electric Company for their able assistance and advice.

The aircraft and flight control descriptions, drawings, and schematics presented in appendixes B and C are used with the permission of Hughes Helicopter Company. The engine description, drawings, and schematics presented in appendix D are used with the permission of General Electric Company.

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INTRODUCTION

BACKGROUND

1. On 22 January 1973, the United States Army Aviation Systems Command (AVSCOM)* awarded a Phase I engineering development contract to the Hughes Helicopter Company (HHC). The contract required HHC to design, develop, fabricate, and test an advanced attack helicopter (AAH) designated YAH-64. Three test vehicles were fabricated by HHC, one ground test vehicle and two flying prototypes. The YAH-64 made its first flight on 30 September 1975. The United States Army Aviation Engineering Flight Activity (USAAEFA) was tasked to conduct Development Test I (DT I) in accordance with the approved test plan (ref 1, app A). The two prototype YAH-64's were delivered to Edwards Air Force Base for the Government Competitive Test (GCT) on 14 June 1976.

TEST OBJECTIVES

2. The objectives of DT I were as follows:

a. To provide engineering data to the AAH Source Selection Evaluation Board (SSEB) for comparison with the AAH systems specification (ref 2, app A).

b. To provide engineering data for determining compliance with the applicable paragraphs of the HHC systems specification for the YAH-64 (ref 3, app A).

c. To provide airworthiness data as a basis for updating the safety-of-flight release (SOFR) for Operational Test I (OT I).

DESCRIPTION

3. The YAH-64 is a two-place, tandem-seat, twin-engine helicopter with four-bladed main and antitorque rotors and conventional landing gear. The helicopter is powered by two prototype General Electric YT700-GE-700 turboshaft engines. The YAH-64 incorporates a T-tail empennage with the fixed horizontal stabilizer mounted above the tail rotor. A 30mm gun is mounted on a turret assembly on the underside of the fuselage below the forward cockpit and provisions are made for mounting 2.75-inch folding fin aerial rockets (FFAR) and tube-launched, optically-tracked, wire-guided (TOW) missile launchers on wing mounting stations. Mission gross weight was calculated to be 14,242 pounds. A detailed description of the aircraft and the flight control systems is contained in appendixes B and C, respectively. Appendix D contains a detailed description of the YT700-GE-700 engines used during these tests.

*Since redesignated the Army Aviation Research and Development Command (AVRADCOM).

TEST SCOPE

4. Flight testing for DT I was conducted at Edwards Air Force Base, California (2303-foot elevation), and at Bishop (4120-foot elevation) and Coyote Flats (9500-foot elevation), California, from 9 July through 30 September 1976. Two instrumented test aircraft were used during this evaluation: Army serial number (SN) 74-22248, the primary handling qualities aircraft; and Army SN 74-22249, the primary performance aircraft. A total of 87 flights for 92.2 hours (62 productive hours) were flown on the two helicopters. Pilots from the Operational Test and Evaluation Agency (OTEA) flew as copilots as often as possible to prepare for OT I. Hughes Helicopter Company installed, calibrated, and maintained the test instrumentation and performed all aircraft maintenance during the test. Flight limitations contained in the SOFR (ref 4, app A) were observed during the evalution. All handling qualities, vibration, and performance items were evaluated against the applicable paragraphs of the systems specification. The vertical agility of the helicopter was evaluated based on a modified vertical displacement maneuver, which is defined in reference 5.

5. The tests were conducted in three external wing store configurations: clean (no external stores or pylons installed); 8-TOW (two TOW launchers installed on each inboard wing store station); and 76-rocket (heavy Hog) (one M200A1 rocket pod on each of the four wing store stations). The 30mm gun was in the straight-ahead stowed position for all tests except weapons firing. Performance, handling qualities, and vibration test conditions are detailed in tables 1 and 2.

TEST METHODOLOGY

6. Established flight test techniques and data reduction procedures were used (refs 6 through 8, app A). Test methods are briefly described in the Results and Discussion section of this report. A Handling Qualities Rating Scale (HQRS) (fig. 1, app F) was used to augment pilot comments relative to handling qualities. Flight test data were obtained from sensitive calibrated test instrumentation and were recorded on magnetic tape. Standard ship's system indicators and calibrated test instrumentation were displayed to the pilot and flight test engineer. Real time telemetry was used to monitor selected critical parameters during certain tests. A detailed listing of the test instrumentation is contained in appendix E. Data analysis methods are described in appendix F.

Table 1. Performance Test Conditions.[1]

Type of Test	Gross Weight (lb)	Longitudinal Center of Gravity[2]	Density Altitude (ft)	Trim Airspeed	Configuration
Hover[3]	13,500 to 17,900	Fwd	4000 to 5620	Zero	Clean[4]
Vertical climb	14,400 to 15,700	Fwd, mid, and aft	3490 to 5980	Zero	76-rocket (heavy Hog) (4 M200 rocket pods)
Forward flight climb	13,600 to 15,300	Aft	[5]2500 to 9000	73 KCAS	8-TOW[6]
Level flight[3]	14,320 to 16,820	Fwd and aft	4720 to 9980	55 to 147 KTAS	Clean and 8-TOW
Vertical displacement	14,180	Mid	5500	140 KTAS	8-TOW
Lateral acceleration	14,200	Mid	5440	Zero	8-TOW
Autorotational descent[7]	13,700 to 14,200	Aft	5000 to 5850	54 to 96 KTAS	8-TOW

[1]100 percent rotor speed unless otherwise noted.
[2]Longitudinal cg: Fwd - 200.0 to 202.3 inches; mid - 202.4 to 204.7 inches; aft - 204.8 to 207.0 inches. All lateral cg's were mid (-0.1 to 0.1 inch).
[3]98 to 100 percent rotor speed (283 to 290 rpm).
[4]Clean configuration: No external stores, wing pylons removed.
[5]Pressure altitude.
[6]8-TOW configuration: One pylon-mounted, faired TOW missile launcher assembly on each inboard wing stores station. Each launcher assembly had the capacity for 4 TOW missiles. Missile openings were plugged to simulate installed TOW missiles.
[7]94 to 104 percent rotor speed (272 to 301 rpm).

Table 2. Handling Qualities and Vibration Test Conditions.[1]

Type of Test	Calibrated Airspeed (kt)	Flight Condition	Density Altitude (ft)	Gross Weight (lb)	Center of Gravity Location (in.)
Static longitudinal stability	48 to 125	Level	8630	14,260	203.6
	62 to 113	Level[3]	6930 to 6590	14,030 to 14,650	206.9
	76	Autorotation	6590	14,480	206.8
	76	Climb	6800	14,440	
Static lateral-directional stability	62 to 113	Level[3]	4800 to 5550	14,140 to 14,650	207.0
	76	Climb	6880	14,170	206.9
	75	Autorotation	6760	14,320	
Maneuvering stability	60 to 117	Turns	6540 to 7820	13,320 to 14,230	206.8
		Pushover and pull-up	6540 to 7460	13,320 to 13,600	206.9
Dynamic stability	60 to 110	Level[3]	5440 to 6580	14,220 to 15,050	206.8 to 207.0
Controllability	56 to 110	Level[3]	5110 to 6240	13,560 to 14,510	206.8 to 207.0
	Zero	Hover[3]	6240 to 11,080	14,050 to 14,360	200.0 to 207.0
Low-speed forward and rearward flight	42 rearward[4]	Level	4090 to 11,094	13,980 to 14,810	199.6 to 200.1
	49 forward[4]				
Sideward flight	44 left[4]	Level	2700 to 11,240	14,190 to 14,650	199.7 to 200.8
	39 right[4]				
Mission maneuvers	Hover to V_{NE}	Hover and level flight	5000	14,400	206.8
Weapons firing	Hover, 50, 90	Hover and level flight	5000	14,400	203.6
Instrument/night flight	Zero to 120	Level	6000	14,400	207.0
Stability augmentation system failures	117	Level	5960	14,520	206.8
Single-engine failures and autorotational entries	80	Level	5640 to 7700	14,300 to 14,500	206.8 to 206.9
	76	Climb	6100	14,420	206.7
Vibration	50 to 135	Level	4720 to 7400	13,650 to 15,600	200.0 to 206.8

[1]Rotor speed 100 percent (290 rpm), SAS ON, unless otherwise noted.
[2]Longitudinal cg: Fwd - 200.0 to 202.3 inches; mid - 202.4 to 204.7 inches; aft - 204.8 to 207.0 inches. All lateral cg's were mid (-0.1 inch).
[3]SAS ON and OFF.
[4]KTAS (knots true airspeed).

RESULTS AND DISCUSSION

GENERAL

7. Performance and handling qualities were evaluated at Edwards Air Force Base (elevation 2303 feet), Bishop (elevation 4120 feet), and Coyote Flats (elevation 9500 feet, handling qualities only). Two instrumented test aircraft were used. The performance evaluation included free flight and tethered hover, vertical climbs, forward flight climbs, level flight, autorotational descent, lateral acceleration, and vertical displacement. The YAH-64 failed to meet three of the six performance requirements of the Army systems specification. It met the requirements of the modified vertical displacement maneuver. Left lateral accelerations were prohibited by the SOFR. The mission gross weight of the YAH-64 was determined to be 14,242 pounds, 1024 pounds in excess of the HHC systems specification weight. Handling qualities of the YAH-64 were satisfactory for the attack helicopter mission; however, three handling qualities deficiencies require correction: the loss of directional control in left sideward flight between 25 and 30 KTAS, tail rotor horsepower exceeding the limit in right hovering crosswinds, and the lack of preset frequencies on the UHF and VHF radios. One shortcoming which had an adverse effect on mission maneuvers, particularly nap-of-the-earth (NOE) flight, was the restricted field of view due to canopy structure and distortion. The low level of physiologically significant vibration of the crew stations was an enhancing feature of the YAH-64. A human factors evaluation of the pilot cockpit revealed four enhancing features and 21 shortcomings; however, the cockpit was a significant improvement in attack helicopter cockpit design. A night lighting evaluation revealed minimal interior reflection of lights from either internal or external sources. The reliability and maintainability of the YAH-64 was adversely affected by four deficiencies: the unreliable auxiliary power unit (APU) starting system, the unreliable main engine pneumatic starting system, the tail rotor drive shaft-mounted cooling fan, and the zero shift of the Marconi fuel quantity gauges. Engine response characteristics did not limit the performance of any maneuver performed during the DT I and OT I phases of this evaluation. Numerous envelope limits were imposed during this evaluation which would be unacceptable for an operational aircraft. A total of 16 enhancing characteristics, 7 deficiencies, and 64 shortcomings were identified. The YAH-64 failed to meet the requirements of 11 paragraphs of the Army systems specification.

PERFORMANCE

General

8. Performance flight testing of the YAH-64 helicopter was conducted at Edwards Air Force Base (elevation 2303 feet) and Bishop (elevation 4120 feet) using aircraft SN 74-22249. The test aircraft was equipped with specially calibrated engines and was instrumented to make in-flight measurements of installed engine intake and exhaust losses. These measured losses were used in conjunction with the General Electric YT700-GE-700 engine specification computer deck to evaluate

aircraft performance in relation to the Army systems specification. Data were recorded on magnetic tape. All forward flight performance tests were flown in coordinated (ball-centered) flight. Test conditions are outlined in table 1 and data analysis techniques are contained in appendix F. This performance evaluation included the following tests: free flight and tethered hover, vertical and forward flight climbs, level flight performance, vertical displacement and right lateral acceleration maneuvers, and autorotational descent performance. All performance measurements were made with the Hughes infrared radiation suppressor system installed; therefore, all performance data includes these losses. The magnitude of these suppressor losses was approximately 16 shp per engine due to exhaust backpressure and an estimated 70 shp required to operate the tail rotor drive shaft-mounted cooling fan. Insufficient data on compressibility effects were gathered near the specification conditions to allow accurate determination of compressibility corrections; therefore, no compressibility corrections were made in performance calculations. All calculated performance data presented in this report were corrected for 0.9 square foot (ft^2) equivalent flat plate drag due to instrumentation. The mission gross weight as defined in paragraph 3.2.2.1.5 of the systems specification was determined to be 14,242 pounds, 1024 pounds in excess of the HHC systems specification weight.

9. The performance of the YAH-64 is compared with the requirements of the Army systems specification in table 3. Of the six performance requirements of the systems specification which were checked, the YAH-64 met or exceeded three. In addition, the YAH-64 met the modified vertical displacement requirements specified in reference 5, appendix A.

Hover Performance

10. Hover performance testing was accomplished at the 2303- and 4120-foot test sites using the tethered and free flight hover techniques with target wheel heights of 5 feet in ground effect (IGE) and 100 feet out of ground effect (OGE). A cable angle indicator was used to maintain the cable vertical. A cable tensiometer measured cable tension as power was changed incrementally from power required for minimum cable tension to topping power. The tests were conducted within a rotor speed range of 98 to 100 percent (284 to 290 rpm). Hover test results are presented in figures 1 through 4, appendix G.

11. The 35°C hot day OGE hover ceiling at the design gross weight of 14,242 pounds was 5350 feet. At 4000 feet, 35°C conditions, the OGE hover maximum gross weight was 15,000 pounds at intermediate rated power (IRP).

Table 3. Performance Specification Compliance.[1]

Test	Specification Requirement	YAH-64 Performance
OGE hover at IRP[2]	NA	15,000-lb gross wt
Vertical climb rate at 95 percent IRP[2,3,4]	450 to 500 ft/min	184 ft/min
Level flight cruise airspeed at MCP[2,3,4,5]	145 to 175 KTAS	141 KTAS
Single-engine level flight airspeed at IRP[2,3,4,5]	90 KTAS	97.5 KTAS
Single-engine service ceiling assuming 95°F ambient temperature[2,3,4,5]	5000 ft	Left engine 4750 ft, right engine 4650 ft
Specification mission endurance with full fuel at sea-level, standard-day conditions[4,5]	2.5 hr	2.71 hr
Lateral acceleration to 35 KTAS[6]	0.25g	0.3g

[1]Army systems specification.
[2]Performance computed at 4000 feet pressure altitude, 95°F ambient temperature, using YT700-GE-700 specification engine data and measured inlet and exhaust losses.
[3]Computed using primary mission gross weight of 14,242 pounds.
[4]No corrections for compressibility were applied, since insufficient compressibility data were generated to establish the magnitude of compressibility effects.
[5]Data corrected for 0.9 ft^2 flat-plate equivalent drag due to instrumentation. Rotor efficiency assumed to be 100 percent.
[6]Lateral acceleration was tested to the right only. Left lateral accelerations were prohibited by the SOFR. Test day condition uncorrected.

Climb Performance

Vertical:

12. Vertical climb performance testing was accomplished at the 2303- and 4120-foot test sites. An Elliott low-airspeed sensor was used to maintain zero longitudinal and lateral airspeed during the maneuver. These climbs were flown in winds of 3 knots or less to minimize the effect of wind gradients during the climb and to assure minimum influence of turbulence, minimum tracking error for ground instruments, and a constant ground reference for the radar altimeter. The aircraft was tested in the 76-rocket (heavy Hog) configuration (four M200 rocket pods) to allow weight to be added after each climb to balance fuel burnoff. Tests were conducted by establishing the aircraft in an OGE hover and then applying incremental power to climb with zero longitudinal and lateral airspeed. Vertical distance data were recorded from a radar altimeter and ground-mounted recording optical instruments (ROI's). The data were correlated with vertical rate data from the vertical speed sensor of the Elliott low-airspeed system. Data from vertical climb performance tests are presented in figures 5 through 8, appendix G.

13. The calculated hot day vertical climb performance is shown in figure 5, appendix G. At a pressure altitude of 4000 feet, 35°C, 95 percent IRP, and mission gross weight of 14,242 pounds, the aircraft could climb vertically 184 feet per minute (ft/min). This fails to meet the requirement of paragraph 3.2.1.1.1.1a of the Army systems specification, which specifies a vertical rate of climb of 450 to 500 ft/min. The maximum gross weight at which the aircraft could meet the minimum vertical rate of climb was 13,880 pounds.

Forward Flight:

14. Continuous single-engine climbs were conducted from near the surface at the 2303-foot test site to the service ceiling (altitude at which a 100-ft/min rate of climb can be achieved) to determine the aircraft's forward flight climb capability and associated Army systems specification compliance. Two continuous single-engine climbs to service ceiling were performed, one on each engine. During the climb, the operating engine was maintained at topping power and main rotor speed was maintained at 100 percent (289 rpm). The best rate of climb airspeed schedule was determined from level flight data. Additional climbs were conducted to determine correction factors for variation in power (Kp) and gross weight (Kw). The climb data were corrected to 35°C hot-day conditions at the design gross weight and to engine model specification power corrected for installation loss. Test results are presented in figures 9 through 11, appendix G.

15. At the mission gross weight of 14,242 pounds, the single-engine service ceiling on a 35°C hot day was 4750 feet on the No. 1 (left) engine and 4650 feet on the No. 2 (right) engine. This failed to meet the Army systems specification requirement of paragraph 3.2.1.1.1.3b of 5000 feet by 250 and 350 feet, respectively.

Level Flight Performance

16. Level flight performance tests were conducted to determine the velocity for minimum power required ($V_{min\ pwr}$), specific range, endurance, and engine performance characteristics. Data were obtained in stabilized coordinated (ball-centered) level flight at incremental airspeeds from approximately 55 knots to the maximum airspeed for level flight (V_H). A constant ratio of gross weight to air density ratio (W/σ) was maintained by increasing altitude as fuel was consumed. Seven level flight performance tests were conducted in the 8-TOW configuration. Five flights were flown to acquire data to generate the basic nondimensional level flight performance parameters at a forward cg. At the request of the SSEB, two additional flights were flown at an average gross weight of 14,400 pounds and an average density altitude of 4720 feet. These tests were to check the effects of changing cg from forward to aft, and to gather performance data at higher airspeeds to account for the increase in the airspeed envelope provided by revision of the SOFR. Three level flight performance tests were flown in the clean configuration to determine the drag of the wing-mounted TOW missile launchers and pylons. The calculated performance data presented in this report were corrected for the 0.9 ft^2 increase in equivalent flat plate area (Δf_e) caused by the test instrumentation mounted externally on the aircraft. The results are presented nondimensionally in figures 12 through 14, appendix G, and dimensionally in figures 15 through 32. The nondimensional data were obtained at an average referred main rotor speed of 286.5 rpm.

17. The mission endurance of the YAH-64 at sea-level, standard-day conditions was 2.71 hours. This is within the 2.5 to 2.8 hours required by the Army systems specification. A breakdown of the sea-level endurance mission fuel load is shown in table 4.

18. The increase in Δf_e for the 8-TOW external stores configuration was a constant 1.4 ft^2 at all thrust coefficients and airspeeds tested. The level flight cruise airspeed (V_{cruise}) as defined by paragraph 3.2.1.1.1.1b of the Army systems specification at maximum continuous power (MCP) was 141 KTAS (no correction for compressibility effects) (para 19). This is 4 KTAS slower than the minimum requirement of 145 KTAS. The single-engine V_H as defined by paragraph 3.2.1.1.1.3a at IRP was 97.5 KTAS. This is 7.5 KTAS greater than the 90 KTAS requirement. The increase in drag from forward to aft longitudinal cg, as shown by a comparison of figures 19 and 29, appendix G, is very small and could not be determined in this evaluation.

Table 4. YAH-64 Maximum Endurance at Sea-Level, Standard-Day Conditions.[1]

Item	Time (min)	Flight Condition	Fuel Flow (lb/hr)	Fuel Used (lb)
1	8	Maximum continuous power	1188	158
2	16	OGE hover at takeoff gross weight[2] with mission stores	877	234
	16	OGE hover at takeoff gross weight minus one-half expendable ordnance	847	226
3	15	150 KTAS at takeoff gross weight with mission stores	1110	278
	15	150 KTAS at takeoff gross weight minus one-half expendable ordnance	974	244
4	15	Maximum endurance airspeed at takeoff gross weight with mission stores	714	178
	15	Maximum endurance airspeed at takeoff gross weight minus *one-half expendable ordnance*	696	174
5	10	80 KTAS at *takeoff* gross weight with mission stores	717	120
	10	80 KTAS at takeoff gross weight minus one-half expendable ordnance	707	118
6	30	Maximum range airspeed less mission expendable ordnance and fuel used through item 5	907	454
7	12.4	150 KTAS at takeoff gross weight minus one-half expendable ordnance	907	201
Total	162.4 (2.71 hours)			2386

[1]Data corrected for instrumentation drag ($\Delta f_e = 0.9\ ft^2$); fuel flow based on 5 percent conservatism (*ie*, fuel flow 105 percent of specification fuel flow); takeoff gross weight is 14,998 pounds including full internal fuel.
[2]Weight buildup in table 5 was used to compute takeoff gross weight, with the exception that full fuel load (2386 pounds) was used in place of the mission fuel load shown in table 5.

19. The high thrust coefficient flights were (C_T x 10^4 = 99.5) flown at average rotor speeds of 284 and 290 rpm (98 and 100 percent) to investigate compressibility effects. Results are compared in figures 26 and 27, appendix G. The higher rotor speed data shown in figure 27 had an average advancing tip Mach number increase of 0.014 when compared to figure 26. The dashed line through the data points on figure 27 show a power rise with increasing airspeed above 90 KTAS compared to the carpet plot data. The point of power divergence began at an advancing tip Mach number of 0.8 and increased to 80 shp at 130 KTAS. These data indicate that at high values of C_T compressibility is a significant factor in the level flight performance of the YAH-64 and should be further evaluated during follow-on testing. Insufficient data were obtained near the systems specification conditions to correct the MCP level flight cruise airspeed for compressibility effect. The data presented in figure 27 is for a C_T x 10^4 of 99.5 and a referred rotor speed of 295 rpm. The Army systems specification conditions require a C_T x 10^4 of 77.5 and a referred rotor speed of 280 rpm. Both the lower C_T and the lower referred rotor speed of the specification conditions would reduce the compressibility effects below the 4 KTAS value shown in figure 27.

20. The mission gross weight is the operating weight from the SOFR (ref 4), plus the primary mission payload as defined in paragraph 3.2.2.1.5 of the Army systems specification and the primary mission fuel load as defined in paragraph 3.2.1.1.1.1c. The mission gross weight is itemized in table 5. The primary mission payload consists of 8-TOW missiles and 728 pounds of 30mm ammunition. A breakdown of the primary mission fuel load is shown in table 6. The mission gross weight was determined to be 14,242 pounds, 1024 pounds in excess of the HHC systems specification weight.

Vertical Displacement

21. The vertical displacement maneuver (fig. A) was evaluated at the 4120-foot test site in the 8-TOW configuration. The handling qualities portion of the evaluation is discussed in paragraph 92. Surface winds were 3 knots or less for this evaluation. The vertical displacement maneuver consisted of a constant-heading level flight entry at 140 KTAS, followed by a rapid pull-up to simulate an obstacle or terrain-avoidance maneuver. The requirement of reference 5, appendix A, called for a 200-foot vertical displacement within 1100 to 1300 feet of horizontal distance traveled without losing more than 30 KTAS from the entry airspeed of 140 KTAS.

22. The vertical displacement maneuver was flown collective-fixed with a rapid aft cyclic control input used to achieve a target load factor. A build-up in normal acceleration was conducted to a peak normal acceleration of 1.8g. The results of this evaluation are presented in figure 33, appendix G. The aircraft could accomplish a 200-foot vertical displacement within 1050 feet horizontal distance measured from the point of control input with only a 28-KTAS loss of airspeed. The vertical displacement performance met the requirements specified in paragraph 21.

Table 5. YAH-64 Mission Gross Weight.

Item	Weight (lb)
Weight empty per SOFR	10,495
Unusable fuel per SOFR	38
Engine oil per SOFR	26
Crew	500
Fixed useful load per SOFR (8-TOW missile tubes, two TOW launchers, gun, and two stores pylons)	497
Operating weight	11,556
Primary mission payload (expendable ordnance)	1,056
8-TOW missiles (328 lb) 30mm rounds (728 lb)	
Primary mission fuel for 1.9 hours	1,630
Mission gross weight	14,242

Table 6. YAH-64 Primary Mission Fuel Determination.[1]

Item	Time (min)	Flight Condition	Fuel Flow (lb/hr)	Fuel Used (lb)
1	8	Maximum continuous power	923	123
2	19	90 KTAS at design gross weight with mission stores	699	221
	19	90 KTAS at design gross weight minus one-half expendable ordnance	683	216
3	3	150 KTAS at design gross weight with mission stores	990	50
	3	150 KTAS at design gross weight minus one-half expendable ordnance	984	49
4	16	Hover OGE at design gross weight	1098	293
	16	Hover OGE at design gross weight minus one-half expendable ordnance	1056	282
5	30	Reserve fuel at maximum range airspeed at design gross weight minus expendable ordnance and fuel burn-off in items 1 through 4	791	396
Total	114 (1.9 hours)			1630

[1]Data corrected for instrumentation drag ($\Delta f_e = 0.9\ ft^2$) (100 percent rotor efficiency assumed); fuel flow based on 5 percent conservatism (fuel flow 105 percent of engine specification fuel flow).

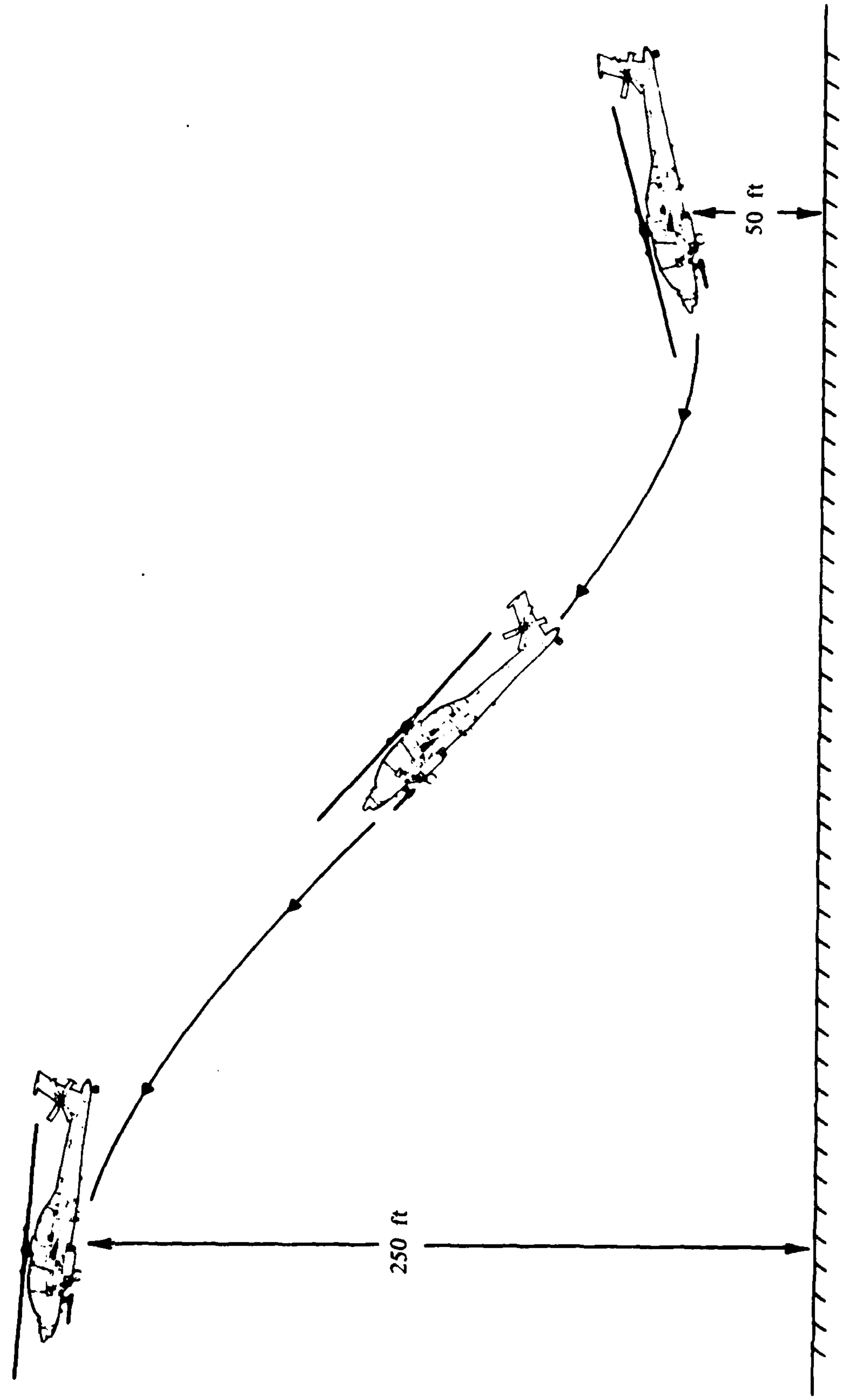


Figure A. Vertical Displacement Maneuver.

Lateral Acceleration

23. Right lateral accelerations (fig. B) were performed at the 4120-foot test site. The handling qualities portion of this evaluation is presented in paragraph 91. Rapid left lateral accelerations were prohibited by the SOFR due to the loss of directional control in 25 to 30-knot left sideward flight (para 77). The lateral acceleration performance was evaluated by conducting lateral accelerations from OGE vertical climb by rapidly rolling the aircraft to a predetermined target roll attitude, maintaining a constant altitude. Directional control was used as necessary to maintain a constant heading during the acceleration. The test was conducted at roll attitudes up to the maximum angle at which constant heading could be maintained using limit tail rotor horsepower (675 shp) (ref 4, app A). Tail rotor horsepower was monitored at the pilot instrument panel using test instrumentation (app E). Performance data were recorded by on-board instrumentation and ground-mounted space positioning equipment. Surface winds were less than 3 knots. The variation of lateral acceleration with average roll attitude is presented in figure34, appendix G.

24. The maximum roll attitude in right sideward flight was 14 degrees and was restricted by a tail rotor horsepower limit. This limit was repeatedly reached without full left pedal deflection being achieved. Maximum average lateral acceleration achieved in right sideward flight was 0.3g, with 6.2 seconds needed to reach 35 knots. This exceeds the requirements of the Army systems specification (0.25g to 35 knots) by 0.05g.

Autorotational Descent Performance

25. Tests were conducted to determine the autorotational descent performance of the YAH-64 helicopter in the 8-TOW configuration. To determine the airspeed for minimum rate of descent (V_{min} R/D), rotor speed was held constant at 100 percent (289 rpm) and data were obtained at stabilized airspeeds in steady-state descents from 50 to 87 knots calibrated airspeed (KCAS). Another series of autorotational descents was conducted at V_{min} R/D by varying rotor speed from approximately 273 to 301 rpm (94 to 104 percent) in approximately 5-rpm increments. The results of these tests are presented in figures 35 and 36, appendix G.

26. The minimum autorotational rate of descent was 1969 ft/min at 71 KCAS. As shown in figure 35, appendix G, airspeed variations from approximately 61 to 81 KCAS resulted in a change in rate of descent of only 80 ft/min, which indicates that airspeed can vary considerably from V_{min} R/D without significantly increasing the rate of descent. At the maximum autorotational airspeed tested (94 KCAS) the rate of descent was 2043 ft/min, resulting in a glide ratio of 4.54:1. As shown in figure 36, the optimum rotor speed during steady-state autorotational descent was 290 rpm (100 percent) with a 1969-ft/min rate of descent. The maximum variation in rate of descent over the rotor speed range for the 8-TOW configuration was 240 ft/min or less, indicating that rotor speed can vary from

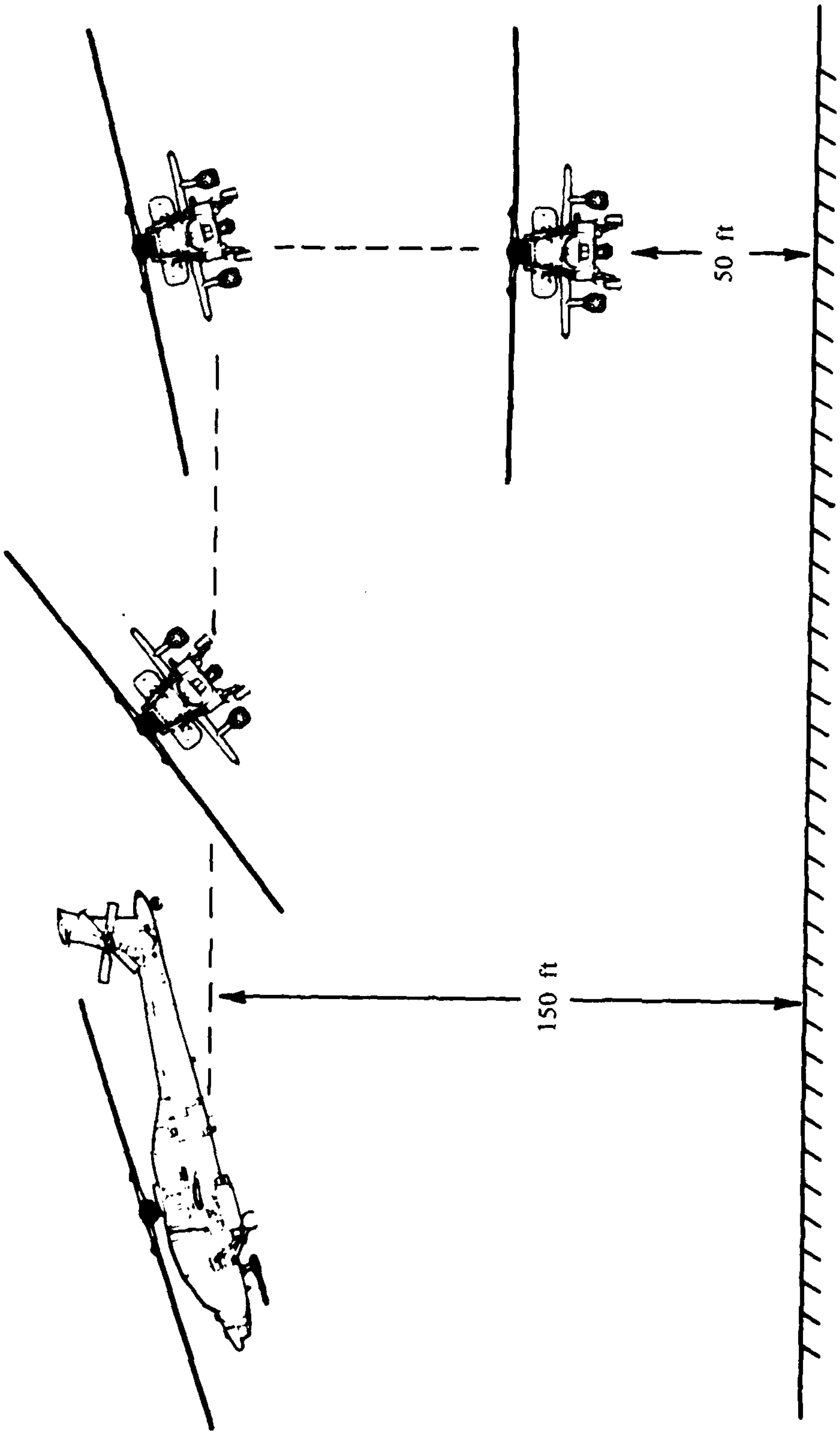


Figure B. Lateral Acceleration Maneuver.

the rotor speed for minimum rate of descent without significantly increasing the rate of descent. The steady-state autorotational descent performance characteristics are enhancing, since moderate deviations from the optimum airspeed and rotor speed conditions do not significantly affect autorotational descent performance.

HANDLING QUALITIES

General

27. Handling qualities tests were performed at Edwards Air Force Base (elevation 2303 feet), Bishop (elevation 4120 feet), and Coyote Flats (elevation 9500 feet). The trim conditions for handling qualities tests were wings-level coordinated (ball-centered) flight. Handling qualities test conditions are listed in table 2. The handling qualities evaluation encompassed both engineering flight test maneuvers and operational-type flying. An HQRS was used to convey the degree of difficulty or pilot effort required to accomplish specific tasks.

28. The YAH-64 exhibited nine enhancing characteristics attributable to handling qualities. The aircraft handling qualities with SAS OFF were excellent. SAS hardover failures were quickly alleviated by the excellent SAS monitor system. Torque matching between the two engines was excellent throughout the engine torque range. Simulated target engagements were enhanced by the ability of the YAH-64 to rapidly attain and precisely maintain hover altitude. The tail wheel configuration of the aircraft was well-suited to operation from unimproved areas since it allowed landing in a decelerating attitude and provided protection from inadvertent ground contact while flying NOE. The capability to taxi rearward enhanced the ground handling task. The automatic radio transmitter selecter system (ARTSS), controlled by a button on the pilot cyclic stick, allowed the pilot to individually select each of three radio transmitters without removing his hands from the controls. This greatly facilitated communication while flying NOE. The flashing caution panel effectively emphasized aircraft system problems. The excellent rotor speed control in autorotational descent greatly decreased pilot effort in maintaining autorotational rotor speed during autorotation.

29. Three deficiences in the handling qualities of the YAH-64 were identified and each is discussed below. The loss of directional control in 25- to 30-KTAS left sideward flight (left hovering crosswind) resulted in a 10- to 45-degree uncommanded yaw excursion. Tail rotor horsepower exceeding the limits was required to hover with right crosswinds gusting to 20 knots. The lack of preset frequencies on the VHF and UHF radios is a deficiency in NOE flight and in simulated instrument meteorological conditions (IMC). Numerous envelope limits were imposed during this evaluation which would be unacceptable for an operational aircraft.

30. Nineteen shortcomings were identified as a result of handling qualities testing. The pilot field of view was restricted by canopy structure and distorted by the windshield and the copilot side canopy panels. This shortcoming was particularly evident in NOE flight and night landings. The pilot field of view was further

restricted by the nose-high attitude during high power climbs, steep approaches, and quick stops. Aircraft flying qualities in simulated IMC were satisfactory; however, a pitch-up which occurred between hover and 60 KIAS, requiring a 3-inch forward cyclic input to maintain aircraft acceleration, degraded flight path accuracy during simulated instrument takeoffs (ITO's). The static stability characteristics degraded aircraft trimmability when extremely precise trim tasks were attempted; however, once established, the trim condition could be easily maintained. Maneuvering stability was positive, although constant-airspeed turns in excess of 45 degrees bank angle were more difficult to coordinate due to a pitch-to-sideslip coupling. Pilot workload during low-speed maneuvering tasks was increased by longitudinal control shifts caused by power variations and by left crosswinds above 15 knots. Dynamic stability characteristics were generally deadbeat, which reduced pilot compensation required to maintain flight path accuracy. Aircraft responses to simulated single-engine failures from dual- and single-engine flight were mild, and adequate control was maintained in autorotation down to the lowest rotor speed tested (80 percent, 231 rpm). Weapons firing tests conducted with both the 30mm gun and 2.75-inch FFAR revealed minimal aircraft reactions which were easily damped by the pilot. Weapon gases from both weapons tested were annoying to both crewmembers. Weapon flash characteristics during night firing were satisfactory. Seven items of noncompliance with the requirements of the Army systems specification were noted.

Control System Mechanical Characteristics

31. The control system mechanical characteristics of the YAH-64 were measured on the ground with the APU operating, force trim system (FTS) ON, and force feel system (FFS) ON and OFF and in flight throughout the flight test program. Results are presented in figures 38 through 42, appendix G. Table 7 is a summary of control system mechanical characteristics. Table 8 is a breakdown of system specification requirements.

32. Longitudinal control centering was positive, but not absolute, particularly for aft stick displacements. Longitudinal control centering was considerably better for forward stick displacements than for aft, an annoying characteristic. Poor longitudinal cyclic centering after aft control displacement is a shortcoming. Lateral centering was positive, and almost absolute; directional centering was positive.

33. Control jump (an undesirable jump in control position when retrimming) in the longitudinal and lateral controls was negligible; however, control jump was significant in the directional control axis with activation or deactivation of the FTS. This directional control jump was particularly annoying during hover and ground taxi operations. The directional control jump when retrimming is a shortcoming. The YAH-64 failed to meet the requirements of paragraph 10.3.3.2.2 of the systems specification, in that no control jump on retrimming is permitted.

34. Control forces could readily and easily be trimmed to zero in all three axes. Control force gradients presented in figures 39 through 42, appendix G, were measured with the FTS ON and OFF. All gradients were within those limits

Table 7. Control System Mechanical Characteristics.

Test Parameter	Control System		
	Longitudinal	Lateral	Directional
Breakout force (plus friction) (lb)	1.4 fwd, 1.8 aft[1]	0.6 left, 0.8 right	5.5 left, 3 right
Full control travel (in.)	9.5	9.0	5.9
Control oscillation	None	None	None
Free play (in.)	Negligible	Negligible	Negligible
Mechanical coupling	None	None	None
Force to move stick 0.5 inch from trim (lb)	2.4 fwd, 3.3 aft[1]	0.1 left, 0.7 right[2]	NA
	1.8 fwd, 1.5 aft[2]		
Limit control force (lb)	12.5 fwd, 13.2 aft[1]	3.0 left, 2.5 right[1]	15 left, 10 right[1]
	5.9 fwd, 4.8 aft[2]		
Control centering	Positive	Positive	Positive
Control jump	Negligible	Negligible	Significant
Control forces trimmable to zero	Yes	Yes	Yes
Force gradient (lb/in.)	2.2 fwd, 2.1 aft[1]	0.26 left, 0.33 right[2]	2.5 left, 2.9 right[2]
	0.79 fwd, 0.83 aft[2]		

[1]Force trim system ON.
[2]Force trim system OFF.

Table 8. Control System Mechanical Characteristics Specification Requirements.

Test Parameter	Army Systems Specification	
	Requirement	Paragraph
Breakout force (plus friction) (lb)	0.5 min, 1.5 max, long. and lat	10.3.2.1.1 and 10.3.2.1.2
	1.0 min, 10.0 max dir	
Control oscillation	None permitted	10.3.3.2.2
Free play (in.)	±1.0% control travel	10.3.2.8
Force to move stick 0.5 inch from trim (lb)	Force required for 0.5-inch stick travel must be less than or equal to breakout force (plus friction)	10.3.2.2.1
Limit control force (lb)	20 long., 10 lat, 50 dir	10.3.3.2.1
Control centering	Positive	10.3.2.2
Control jump	None permitted	10.3.3.2.2
Control forces	Trim to zero	10.3.2.5
Force gradient (lb/in.)	See Table II, ref 4, app A	10.3.2.2

specified in table II, reference 2, appendix A. There was no mechanical coupling in any of the controls. The breakout forces (plus friction) were not objectionable, although the aft breakout force (plus friction) failed to meet the requirements of paragraph 10.3.2.1.1 of the Army systems specification by 0.3 pound.

Control Positions in Trimmed Forward Flight

35. Control positions in trimmed forward flight were evaluated throughout the test program. Typical characteristics are shown in figure 43, appendix G. The longitudinal control position gradient with airspeed was essentially stable, with a slightly steeper gradient in the 50- to 65-KCAS range, nearly neutral gradient in the 65- to 90-KCAS range, and a fairly constant stable gradient out to 125 KCAS. The longitudinal trim change between 50 and 65 KCAS was particularly noticeable during simulated ITO and during NOE flight. This longitudinal trim change was a factor in the shortcoming discussed in paragraph 71. The instantaneous trim release button and the force trim release switch both allowed control forces to be easily and readily trimmed to zero (HQRS 2). Cyclic control travel for the airspeed range from 50 to 125 KCAS was 1.25 to 2 inches, varying with weight and cg. Longitudinal control margins at the test airspeed extremes were adequate. The pilot experienced some difficulty trimming the aircraft longitudinally within 2 knots of desired airspeed in the airspeed range from 50 to 90 KCAS (HQRS 5). Some of the trimming difficulty was attributed to the horizontal stabilizer being in a transitory flow state at approximately 60 KCAS. Steady level flight in this airspeed range is often used during low level flight operations; contour flight operations require the pilot to hold a constant airspeed. The trim difficulty encountered in the 50- to 90-KCAS airspeed range was annoying and is discussed further in paragraph 41.

36. The variation of lateral and directional control positions with calibrated airspeed was minimal. Total lateral cyclic control travel for the 50- to 125-knot range was approximately 1/2 inch with the stick being nearly centered at 125 KCAS. Total directional control travel for the airspeed range was approximately 1/2 inch. Lateral and directional control margins at the test airspeed extremes were adequate. The instantaneous force trim release button and the force trim release switch both allowed lateral and directional control forces to be easily trimmed to zero (HQRS 2). The lateral and directional trim control position characteristics of the YAH-64 are satisfactory.

37. The variation of pitch attitude with calibrated airspeed showed a noticeable increase in pitch attitude with increased airspeed (negative gradient) in the 50- to 60-KCAS airspeed range. This was particularly noticeable in the aft cg configuration. The pitch attitude gradually became positive in the 60-KCAS area as the horizontal stabilizer became effective, and remained positive as airspeed was further increased to 125 KCAS. The increase in pitch attitude with increasing airspeed from 50 to 65 KCAS failed to meet the requirements of paragraph 10.3.4.1.2 of the Army systems specification.

Static Longitudinal Stability

38. Static longitudinal stability characteristics were evaluated at the conditions listed in table 2. The aircraft was trimmed in wings-level ball-centered flight at the desired airspeed. With the collective control fixed, airspeed was stabilized both faster and slower than the trim airspeed while maintaining trimmed flight. Tests were flown with SAS ON and OFF and with SAS ON but FFS OFF. Data recorded at each stabilized airspeed are presented in figures 44 through 47, appendix G.

39. At all conditions tested, the operating condition of the SAS did not significantly affect the static longitudinal stability gradients. Longitudinal cyclic control position and control force did not provide usable cues to the pilot concerning small deviations (±10 knots) from the trim airspeed at airspeeds above 60 KCAS.

40. The static longitudinal stability of the YAH-64 about a trim airspeed of 62 KCAS showed an extremely stable gradient of longitudinal cyclic control position at airspeeds lower than trim and an essentially neutral gradient at airspeeds higher than trim. A strong positive static stability normally aids in maintaining the trim airspeed of an aircraft; however, this was not true in this instance. As airspeed was increased between 45 and 60 KCAS the pitch attitude became increasingly nose-up. The strong positive static stability and the increasing pitch attitude with increasing airspeed were probably caused by unsteady flow effects on the horizontal stabilizer. The combination of apparent unsteady aerodynamics, the noticeable longitudinal trim shift with increasing airspeed, and the contradictory attitude cue of increasing pitch attitude with increasing airspeed in the 45- to 60-KCAS airspeed range made airspeed control difficult (HQRS 5). The control shift was particularly noticeable during simulated ITO's and NOE flight. The longitudinal cyclic control position change between 45 and 60 KCAS contributed to a shortcoming discussed in paragraph 71.

41. At a trim airspeed of 81 KCAS, the aircraft was statically stable (forward cyclic required to maintain increased airspeed), as indicated by the gradient of longitudinal cyclic control position with airspeed. At 113 KCAS the aircraft was statically unstable (aft cyclic required to maintain increased airspeed). The variation of pitch attitude with airspeed in both cases was stable (nose-down for increased airspeed). The longitudinal static stability characteristics degraded aircraft trimmability at airspeeds greater than 60 KCAS when small airspeed changes (2 to 4 knots) were attempted (HQRS 4) and are a shortcoming. Once a precise airspeed was established it was easily maintained. Although the pitch attitude changes with airspeed variations were small, they were easily detected by the pilot. Pilot workload was decreased during these tests by the gust alleviation and pitch attitude hold characteristics of the automatic stabilization equipment (ASE), which minimized pilot effort to maintain airspeed. While small trim airspeed changes were difficult to precisely trim, large airspeed changes where level flight was maintained did not present a problem.

42. The combined effects of power and vertical speed on static longitudinal stability were evaluated in IRP climbs and autorotational descents. Figure 47, appendix G, shows that static stability in both cases was neutral. Airspeed was difficult to establish during the maximum power climb due to the longitudinal trim shift caused by increased power, the lack of pitch attitude cues, and the nose-high attitude. Once airspeed was established in the climb, it was easily maintained (HQRS 2). The 12- to 13-degree nose-up attitude in forward flight climbs made it impossible to adequately clear the aircraft of other traffic without clearing turns. The restricted forward field of view due to nose-high attitude is a shortcoming. Further mission applications of this shortcoming are discussed in paragraphs 72 and 90. Airspeed was more easily established in the descent.

43. Large longitudinal cyclic control changes were required when making large power changes at a constant airspeed. Transitioning from autorotational descent to topping power required a 2.5-inch forward cyclic input. Large power changes were frequently required when the aircraft was operated in NOE or contour flight in mountainous terrain, and the power variations caused an increased pilot workload in longitudinal cyclic control (HQRS 6). The large longitudinal cyclic trim change due to power variations is a shortcoming, although the requirements of the Army systems specification were met.

44. The static longitudinal stability characteristics of the YAH-64 failed to meet the following requirements of the Army systems specification:

a. Paragraph 10.3.4.1, in that a static longitudinal instability exists about a 113-KCAS trim airspeed.

b. Paragraph 10.2.4.1.2, in that aircraft pitch attitude becomes more nose-up with increasing airspeed between 45 and 60 KCAS.

Static Lateral-Directional Stability

45. Static lateral directional stability characteristics were evaluated at the conditions listed in table 2. Tests were conducted by trimming the aircraft in wings-level ball-centered flight at the desired airspeed. With the collective control fixed and maintaining a steady heading at the trim airspeed, the aircraft was stabilized at incremental sideslip angles left and right of trim. Test results are presented in figures 48 through 51, appendix G.

46. At all conditions tested, the gradient of directional control position versus sideslip angle was linear and indicated positive static directional stability (increased left directional control to maintain increased right sideslip).

47. Positive dihedral effect was indicated by increasing lateral control displacement in the direction of sideslip as the sideslip angle increased. Trends generally showed a stronger dihedral effect in right sideslips than in left sideslips.

48. Side-force characteristics, as indicated by the variation of roll attitude with sideslip, were positive and linear at all airspeeds. Side-force gradients became steeper with increasing airspeed and provided adequate sideslip cues to the pilot.

49. An aerodynamic coupling of pitch attitude response to sideslip was observed in the YAH-64. This coupling is evident in the variation of longitudinal cyclic control position with sideslip angle. At all airspeeds tested, increasing sideslip angles left of the trim sideslip angle required increasing aft cyclic control, indicating a coupling of nose-down pitch response to left sideslip. For a 5- to 7-degree sideslip variation to the right of the inherent sideslip angle at the 62-KCAS trim airspeed, the longitudinal cyclic position was essentially constant. At trim airspeeds of 82 and 113 KCAS, the longitudinal control was displaced forward for 15 to 20 degrees of sideslip to the right of the inherent sideslip angle. This indicates that a coupling of nose-up pitch response to right sideslip exists but is a function of both airspeed and sideslip deviation from the inherent sideslip angle. This pitch-to-sideslip coupling causes problems in turning manuevering flight and was determined to be a shortcoming for maneuvering flight (para 54).

50. The combined effects of power and vertical velocity were evaluated by flying steady-heading sideslips during topping power climbs and autorotational descent. The aircraft exhibited a slightly stronger static directional stability in climbs than in descents. Similarly, side-force characteristics and dihedral effect were more pronounced in climbs. Sideslips to the right of trim sideslip in climbs could not be evaluated due to tail rotor power limits.

51. Cyclic-only turns were conducted during this evaluation. Turns and rolls were always promptly initiated in the proper direction. Cyclic-only turns up to standard rate 3 deg/sec in either direction were possible at all airspeeds; however, these turns were uncoordinated. The strong side-force characteristics prompted the pilot to apply directional control into the turn to achieve a coordinated turn. While cyclic-only turns are possible, a more desirable turn is accomplished by coordinating cyclic and directional control into the turn. Pedal-only turns resulted in the aircraft pitching nose-up in left turns and nose-down in right turns (pitch-to-sideslip coupling), while rolling due to positive dihedral effect. The static lateral-directional handling qualities of the YAH-64 are satisfactory.

Maneuvering Stability

52. Maneuvering stability was evaluated during pushovers and pull-ups and constant-collective, constant-airspeed left- and right-hand turns under the flight conditions listed in table 2. The variation of longitudinal cyclic control position and pitch rates with cg normal acceleration was determined with step inputs up to 2 inches. Additionally, the variation of longitudinal cyclic control position and stick force with cg normal acceleration was determined by stabilizing the aircraft at increasing bank angles, holding airspeed and collective position constant with ball centered during the maneuver. Maneuvering stability characteristics are presented in figures 52 through 57, appendix G.

53. Figures 52 through 54, appendix G, show the stick-fixed (control position) maneuvering stability and the stick-free (control force) maneuvering stability of the YAH-64 during fixed-collective, ball-centered turning flight. Both the control position and control force maneuvering stability at 60 KCAS were positive and essentially linear with gradients of approximately 3.4 in./g and 12.5 lb/g, respectively. The aircraft was in a very steep descending spiral at the high g levels at this low airspeed. Descent rates of about 2000 ft/min and high rates of turn resulted in mild pilot disorientation. At the 98-KCAS trim airspeed the control position maneuvering stability was positive and essentially linear at about 1.4 in./g. The control force stability was positive with the gradient decreasing from 12.5 lb/g at 1g trim to 5.5 lb/g at 1.8g. Turning maneuvering stability at 117 KCAS revealed a distinctly nonlinear characteristic; control force stability was positive with a gradient of 2 lb/g up to 1.5g. Between 1.5 and 1.58g the control force gradient steepened to about 50 lb/g. At 1.58g the FFS malfunctioned and dropped off-line, causing the stick force to return to near zero. This sudden reduction of stick force caused a momentary aft cyclic input, but this was quickly neutralized by the pilot with no perceptible increase in normal load factor. Two repeats of this test on two separate occasions resulted in similar occurrences. The FFS problem was solved by Sperry technicians and the FFS was qualitatively evaluated to the limit load factor prior to the end of DT I testing; however, additional maneuvering stability tests were not flown. Control position stability at 117 KCAS was stable, but the gradient was only 0.4 in./g.

54. Stick force cues were qualitatively judged to be adequate at all airspeeds. The wide variation in gradients experienced indicates that the pilot does not really rely on stick force or stick position cues when maneuvering at levels between 1.0 and 2.0. Roll attitude control was adequate up to peak roll attitudes tested (HQRS 3); however, a slight roll attitude overshoot and lateral pilot-induced oscillation tendency was noted when rolling from wings-level to 45 degrees roll attitude or greater. Airspeed control became increasingly difficult (HQRS 5) at roll attitudes over 45 degrees, due to the pitch-to-sideslip aerodynamic coupling discussed in paragraph 49. The pitch-to-sideslip coupling is a shortcoming.

55. Vibration increases with increasing g were mild but noticeable as an increase in the 4-per-rotor-revolution (4/rev) airframe vibration. No perceptible instrument blurring or control vibrations were noted. The maneuvering stability characteristics of the YAH-64 during turning flight failed to meet the requirements of paragraph 10.3.6.3 of the Army systems specification, in that an objectionable coupling of pitch to sideslip was noted.

56. Control position maneuvering stability during steady symmetrical pull-ups and pushovers was evaluated using standard test techniques. Data are presented in figures 55 through 57, appendix G. Control position maneuvering stability was stable at all airspeeds. The gradients were 2.6 in./g at 60 KCAS, 1.7 in./g at 97 KCAS, and 1.9 in./g at 117 KCAS. No divergent pitch tendencies were noted. The aircraft was fully controllable at the lowest g level tested (0.17g). Normal load factors below 0.17 were not tested due to accessory gearbox oil pressure sensor problems. At low g levels (0.3g and below) numerous caution lights were

illuminated. The accessory gearbox low oil pressure caution light often remained illuminated for several minutes, causing the maneuver to be aborted. In one instance the light remained on and a precautionary landing was necessary. The problems with oil pressure sensors are discussed in paragraph 133. No increase in vibration was noted during the pull-up maneuvers. The maneuvering stability characteristics of the YAH-64 during pull-up and pushover maneuvers are satisfactory.

Dynamic Stability

57. The short-term dynamic stability (gust response) characteristics of the YAH-64 were evaluated at the conditions listed in table 2. The short-term characteristics were investigated in three control axes by making single-axis inputs of various amplitudes which were held for 1 second and then returned to the initial position. Following the pulse, all controls were held fixed until the aircraft motion subsided or until a limit was approached. The short-term lateral-directional characteristics were also evaluated by directional control release from steady-heading sideslips. Long-term longitudinal dynamic response characteristics were evaluated by increasing or decreasing airspeed incrementally from the trim airspeed and then returning the controls to the trim position and observing subsequent aircraft motion. The SAS was disengaged to evaluate short-term response characteristics at 91 KCAS, longitudinal long-term response at 60, 91, and 107 KCAS, and to evaluate the lateral-directional response to releases from steady-heading sideslips. Data are presented in figures 58 through 73, appendix G.

58. The short-term (gust response) characteristics of the YAH-64 were essentially deadbeat with SAS ON or OFF. Figures 58 through 61, appendix G, are typical responses. With SAS ON, the dynamics of the aircraft response show deadbeat short-term characteristics followed by a slow return to trim due to the attitude retention feature of the longitudinal SAS. With SAS OFF, short-term characteristics were essentially deadbeat and there was no tendency to return to the trim attitude. Figures 60 and 61 show typical wing flap response to longitudinal control inputs which occurred at airspeeds.

59. The long-term response characteristics were deadbeat with SAS ON and aperiodic with SAS OFF, except in the instance where low airspeed resulted in apparent stalling of the horizontal stabilizer. Figures 62 and 63, appendix G, show typical long-term responses. With SAS ON, the aircraft returned to trim due to the attitude retention feature of the longitudinal SAS. With SAS OFF, an aperiodic divergence was noted in the direction of the last control movement. This divergence was very slow and posed no handling qualities problem that would limit the aircraft in either visual meteorological conditions (VMC) or IMC.

60. During an investigation of the possible dynamic effects of the horizontal stabilizer, an oscillating longitudinal mode was encountered. This mode was attributed to the same phenomenon which caused the longitudinal trim shift between 45 and 65 KCAS (para 41). A time history of this oscillation is presented in figure 64, appendix G. This mode was found only in a contrived demonstration

flown to further highlight the possible dynamic effects of the horizontal stabilizer. Any handling qualities problems attributable to this oscillating mode are addressed in this report in terms of longitudinal trim shifts required to avoid the motion. The longitudinal dynamic stability characteristics of the YAH-64 are satisfactory.

61. The short-term lateral dynamic characteristics of the YAH-64 were deadbeat with SAS ON. With SAS OFF (figs. 66 and 68, app G), the roll attitude and rate response was oscillatory about a roll convergence mode. The oscillatory mode was convergent with a period of approximately 4 seconds and was effectively fully damped in 14 seconds. The damping was such that the pilot felt no need to manually damp the motion. These short-term lateral characteristics posed no handling qualities problems in VMC or IMC with SAS ON or OFF.

62. The YAH-64 short-term responses to directional control pulses are shown as time histories of directional pulses with SAS ON (figs. 69 and 70, app G). Directional responses were deadbeat. No directional pulses were conducted with SAS OFF; however, directional step inputs for controllability revealed no adverse handling qualities due to dynamic stability. In an attempt to excite a Dutch-roll mode, control releases from steady-heading sideslips were evaluated. Time histories are shown in figures 71 through 73. Yaw and sideslip responses were essentially deadbeat with SAS ON or OFF.

Controllability

63. Controllability tests were conducted at the conditions listed in table 2. *Single-axis step control inputs* of various sizes were applied to the longitudinal, lateral, and directional controls in both directions with SAS ON and OFF. Mechanical control fixtures were used to set the longitudinal and lateral control input sizes. Directional control input size was set by use of a control fixture in conjunction with the front cockpit fore and aft pedal adjustment. The single-axis step inputs were held against the control fixture while the other controls remained in their trim position. Results of the controllability tests accomplished at a hover and in forward flight are presented in figures 73 through 88, appendix G. Results were evaluated in terms of control power (attitude change 1 second after a 1-inch longitudinal or directional input and after 0.5 second following a 1-inch lateral input), control response (maximum angular rate (deg/sec/in.) of control displacement), and control sensitivity (maximum angular acceleration (deg/sec^2/in.)) of control displacement.

64. The longitudinal control response with SAS ON was essentially independent of airspeed, ranging from 7.5 deg/sec at a hover and at 110 KCAS to 11 deg/sec at 60 KCAS. The response was linear, in that forward and aft longitudinal control response characteristics were the same. At 110 KCAS aft control response was 2.5 deg/sec greater than the forward response, a minor nonlinearity which was not apparent in flight. With SAS OFF longitudinal response increased from 7.5 deg/sec at a hover to13.5 deg/sec at 90 KCAS and linearity between forward and aft responses was noted. At 110 KCAS the SAS OFF longitudinal control response was nonlinear (13 deg/sec forward and 17.5 deg/sec aft). The time to

reach 63 percent of maximum pitch rate was 0.4 to 0.5 second, SAS ON, and 0.7 to 1.0 second, SAS OFF. Maximum pitch rates were higher, SAS OFF, at the higher airspeeds but no significant difference was noted at the lower airspeeds or at a hover. Longitudinal control sensitivity was unaffected by SAS operation and ranged from 11.5 deg/sec^2 at a hover to 18 deg/sec^2 at 110 KCAS. A minor nonlinearity at 60 KCAS was observed in the data but was not noted in flight. At 60 KCAS aft control sensitivity was 8.5 deg/sec^2 while forward sensitivity was 11 deg/sec^2. All normal acceleration traces showed normal variations of load factors in response to longitudinal step inputs. Aircraft response was always rapid and in the proper direction. No handling qualities problems were observed. An aerodynamic coupling of roll response to pitch rate (roll to pitch) was observed with SAS OFF and to a lesser degree with SAS ON. Nose-up pitch rate resulted in right roll attitude changes and vice versa. This coupling, though evident throughout the flight regime, did not pose any handling qualities problems. The longitudinal controllability characteristics of the YAH-64 are satisfactory.

65. The lateral control response, SAS ON, was essentially independent of airspeed, ranging from 20 deg/sec at a hover and at 110 KCAS to 24 deg/sec at 90 KCAS. A slight lateral control response nonlinearity was observed at 90 KCAS in that left lateral response was 20 deg/sec and right response was 24 deg/sec. With SAS OFF lateral control response characteristics were essentially linear but varied from 20 deg/sec at a hover and at 110 KCAS to 24 deg/sec at 90 KCAS. The time required to reach 63 percent of the maximum roll rate was 0.4 to 0.5 second and was independent of SAS operation. Maximum roll rates were higher with SAS OFF due to the decreased roll damping. Lateral control sensitivity was relatively independent of airspeed and independent of SAS operation. Lateral control sensitivity ranged from 45 deg/sec^2 in a hover to a maximum of 51 deg/sec^2 at 90 KCAS. Aircraft roll responses to lateral control inputs were rapid and in the proper direction. No roll reversals were noted and no perceptible yaw accompanied the lateral inputs. A tendency to slightly overshoot a desired roll attitude greater than 45 degrees was noted but was easily compensated by the pilot. The lateral controllability characteristics of the YAH-64 are satisfactory.

66. Directional response characteristics were essentially independent of airspeed with SAS ON, varying only ±1 deg/sec from the 17-deg/sec hover response throughout the airspeed range tested. With SAS OFF the directional response was 9 to 10 deg/sec at 60, 90, and 110 KCAS. Directional control sensitivity was also essentially independent of airspeed, varying only 5 deg/sec^2 less than the hover sensitivity throughout the airspeed range. The time to 63 percent of maximum yaw rate at the 9500-foot test site in forward flight was generally 0.5 second. Maximum yaw rate was not attained during hover tests at the 2303-foot test site. The most notable characteristic of directional controllability was the strong dependence on SAS operation. Both response and sensitivity were higher with SAS ON. This is just oppposite to the characteristics observed in the lateral and longitudinal characteristics. This disparity in both directional response and sensitivity was attributed to the directional control augmentation system (CAS), which caused a full 10-percent SAS actuator input in the same direction as the directional control input. This CAS input was normally held for the duration of

the control input. A strong coupling of pitch response to sideslip was observed during these tests. This coupling was discussed in paragraph 44. During hover and low-speed flight involving typical mission maneuvering tasks, the pilot was able to quickly and precisely control yaw rate and attitude within the constraints posed by the left sideward flight directional divergence characteristic (para 77) and the restrictive tail rotor power limits (para 75). The directional controllability characteristics of the YAH-64 are satisfactory.

67. The harmony between lateral and longitudinal control response was poor. Lateral control sensitivity was approximately four times the longitudinal control sensitivity and lateral response was almost three times the longitudinal response. This characteristic was noticed by the pilot on the first takeoff to a hover as a tendency for lateral pilot-induced oscillation. The lateral sensitivity was quickly mastered and presented no problem during the remainder of this evaluation; however, the lack of cyclic control harmony is a shortcoming. The directional control system was found to have a large amount of hysteresis between the forward and aft directional controls. With the copilot pedals hard against a control fixture, it was possible for the pilot to make directional control inputs of up to 1.5 inches. This phenomenon was not apparent in flying the aircraft normally. Tests were not conducted to determine the reason for this directional control hysteresis.

Ground Handling Characteristics

68. Ground handling characteristics (taxiing) were evaluated on a daily basis on concrete ramps and taxi ways with a portion of the taxi way inclined approximately 3 degrees. Wind conditions were generally calm except for occasional conditions with gusts to 25 knots.

69. An initial collective increase to approximately 20 percent and slightly forward cyclic was required to initiate forward movements. Pedal pulses were occasionally required to help unlock the tail wheel due to a binding of the locking pin. The intermittent binding of the tailwheel locking pin is a shortcoming. Acceleration to taxi speeds up to 30 knots required increasing forward cyclic at a constant collective setting. Neither droop stop pounding nor ground resonance was encountered during this evaluation. The aircraft exhibited unstable directional stability due to the tail wheel configuration, but was easily controlled (HQRS 2). Taxi turns of over 45 degrees heading change at normal taxi speeds (approximately 5 to 10 knots) resulted in an outside-wing-down change of 1 to 2 degrees without cyclic compensation. This rolling characteristic was annoying, but could be eliminated by using cyclic into the turn. Wind gusts from any direction did not affect taxiing characteristics. Braking characteristics required excessive pedal pressures (HQRS 4), but provided adequate stopping power at normal taxiing speeds. The excessive brake pedal pressure required is a shortcoming. Additionally, braking characteristics at high ground speeds (up to 45 knots), as after a run-on landing, were poor, resulting in unsatisfactory stopping distances of approximately 300 to 400 feet. The poor braking characteristics at high ground speeds are a shortcoming. The aircraft can be stopped without brakes using coordinated aft cyclic and down collective, but this results in excessive stopping distances.

70. The aircraft could be safely taxied rearward without contacting droop stops or compromising main rotor/tail boom clearance. With differential braking and cyclic control, the aircraft could be easily pivoted about a point between the main landing gear, allowing taxi maneuvering into and out of small areas, such as rearming revetments. The capability of the aircraft to safely taxi rearward is an enhancing characteristic.

Takeoff and Landing Characteristics

71. Takeoff and landing characteristics were evaluated throughout the test at gross weights from 14,400 to 17,400 pounds, forward and aft cg, and SAS ON and OFF. Wind conditions ranged from generally calm to occasionally gusting to 25 knots. Lift-off to a hover at an aft cg resulted in a 2-degree left-wing-down and a 4-degree nose-up attitude change which required minimal pilot compensation (HQRS 3). At a forward cg the nose pitch-up was not apparent. Hover stability was satisfactory with SAS ON and OFF, and hover positions were held easily (HQRS 2). The test hover height of 10 feet was easily held to ±1 foot accuracy (HQRS 2). Field of view from both cockpits was satisfactory. Transitioning to forward flight, the aircraft required a nose-down attitude (zero to 5 degrees nose-down) until approximately 50 to 60 KCAS; the aircraft would then pitch up 4 to 6 degrees. The acceleration from hover to 60 KCAS required an excessive forward cyclic input of approximately 3 inches to maintain the acceleration. The pitch-up at 50 to 60 KCAS under night instrument flight conditions could induce momentary mild spatial disorientation, requiring moderate pilot compensation (HQRS 4) while adjusting to the situation. The pitch-up from hover to 60 KCAS requiring a 3-inch forward cyclic control input is a shortcoming. Running takeoffs (up to 20 knots) required a small aft cyclic input after lift-off, due to a slight pitch-down characteristic. The pitch-down was more apparent at forward cg's, but still required only minimal pilot compensation to execute a smooth takeoff (HQRS 3). Vibrations through translational lift (approximately 25 to 35 KCAS) were mild, at 4/rev frequency.

72. Deceleration from forward flight to a hover was characterized by nose-high attitudes (10 to 15 degrees) during the initial flare, which limited the forward field of view from both cockpits and caused both crewmen to lose sight of the intended landing area. The pilot relied heavily on peripheral vision until a hover was established (HQRS 5). Recovery from the flare required minimal left pedal input to compensate for the power changes (HQRS 3). With SAS OFF, the pedal change introduced a slight tendency for directional pilot-induced oscillation. Landings into confined areas required steep approach angles which further degraded the pilot's view of the landing area. The restricted forward field of view due to nose-high attitude is a shortcoming previously mentioned in paragraph 42.

73. Run-on landings were accomplished at touchdown speeds of up to 45 knots. Forward field of view from both cockpits was significantly better than in approaches to a hover. During landing the tail wheel touched down first, and bounced if the sink rate was greater than approximately 200 ft/min. The main landing gear cusioned these sink rates without inducing any vertical bouncing tendencies.

Smooth, precise run-on landings were easily accomplished (HQRS 2). The run-on landing characteristics are satisfactory.

Low-Speed Flight Characteristics

74. The low-speed flight characteristics of the YAH-64 were evaluated at the conditions listed in table 2. These tests were accomplished to assess low-speed handling qualities and to simulate hovering with a head wind, tail wind, or wind from the right or left side. A ground pace vehicle was used as an airspeed reference during these tests. Surface wind conditions were 3 knots or less. Tests were conducted at a wheel height of 15 feet (IGE). Additional forward and rearward points were flown at a wheel height of 100 feet (OGE). Test data are presented in figures 89 through 92, appendix G.

Sideward Flight:

75. In right sideward flight the increasing right lateral cyclic control position with increasing right sideward velocity was essentially linear and indicated a strong right lateral stability. The variation of longitudinal cyclic control with right lateral airspeed was minimal to the highest airspeed tested. Although a longitudinal control reversal was present at the 9500-foot test site, the total variation in longitudinal cyclic control position in right sideward flight was less than 0.5 inch. This reversal was not objectionable. The variation of left directional control position indicates a distinct but unobjectionable gradient shift at 15 KTAS in right sideward flight. The increasing left pedal required in right sideward flight between zero and 15 KTAS was directly reflected by a corresponding increase in the tail rotor power required. At 15 KTAS right sideward velocity, the tail rotor was operating within the 2-minute limit (369 to 505 shp) gauge with occasional peaks into the 6-second operation limit (505 to 675 shp). As sideward velocity was increased to 19 knots at the 2303-foot test site, left pedal requirement was decreased and tail rotor power requirements decreased. At 28 KTAS right sideward flight, the tail rotor power requirement was below the continuous operation limit (330 shp), but further increases in right sideward velocity again resulted in increased left directional control and tail rotor power required. At the 9500-foot test site, handling qualities in right sideward flight were good and pilot compensation to maintain the desired flight path was not a factor (HQRS 2); however, tail rotor power requirements at 15 KTAS right sideward velocity approached the limits of the present tail rotor drive system. During one OGE hover in winds of 15 knots gusting to 20 knots, the YAH-64 could not be turned through the 90-degree right relative wind angle without exceeding the tail rotor power limits. Tail rotor horsepower exceeding the limit in right hovering crosswinds is a deficiency.

76. In left sideward flight the variation of lateral cyclic control position with increasing left sideward velocity showed an essentially neutral left sideward stability to 30 KTAS. At this airspeed, a directional instability was encountered at the test density altitude of 4210 feet. Although it was possible on one occasion to stabilize the aircraft at 30 KTAS left sideward flight, control was generally not possible at airspeeds greater than 25 KTAS left sideward flight. This instability was not encountered at a density altitude of 11,240 feet. The directional instability in left sideward flight was highly sensitive to relative wind angle. Heading deviations as little as 10 degrees, nose into or out of the 90-degree left relative wind, avoided the divergence. Control was easier when the aircraft was yawed 10 degrees, nose into the relative wind, although control could be maintained with the tail 10 degrees into the relative wind at the expense of increased pilot effort. This phenomenon, which is discussed in paragraph 77, caused a change in test technique at the lower altitude, in that the 30- and 45-KTAS points were flown with the aircraft heading 10 degrees left of the heading for left sideward flight. The variation of longitudinal control position with increasing left lateral airspeed to 30 KTAS required a 1.5-inch aft cyclic control displacement. This longitudinal control shift was not abrupt but was noticeable to the pilot, since the change in cyclic control position made activation of the radio and intercom push-to-talk switch more difficult. The longitudinal control shift required moderate pilot compensation (HQRS 4) in holding a precise pitch attitude with a 15-knot gusting left crosswind. The longitudinal control shift in left sideward flight is a shortcoming. Directional control position in left sideward flight was essentially unchanged from hover to 13 KTAS. Increasing right pedal was required to the limit airspeed of 45 KTAS in modified left sideward flight.

77. The instability in left sideward flight had an onset airspeed and a severity which seemed to be a function of the left sideward acceleration, heading control as the critical airspeed was approached, and density altitude. The instability was judged to be more severe during pilot training at near sea-level conditions, and it was not encountered at 11,240 feet density altitude. The following general discussion of the directional instability applies to the phenomenon as observed at the 4210-foot density altitude. When increasing airspeed gradually from 25 to 30 KTAS to the left and maintaining precise heading control, it was possible to attain 30 KTAS in stabilized left sideward flight. Beyond this condition yaw control could not be maintained and a yaw divergence resulted (HQRS 10). When the transition between 25 and 30 KTAS was attempted more rapidly, it was not possible to attain 30 KTAS in stabilized left sideward flight. The yaw divergence, usually occurring between 25 and 30 KTAS, was manifested by the aircraft yawing 10 to 45 degrees, tail into the relative wind, followed by a nose-down pitch attitude change. The loss of directional control in 25 to 30 KTAS left sideward flight (left hovering crosswind) is a deficiency.

78. Yaw control was recoverable after the initial yaw divergence. Recovery from the divergence was accomplished in two ways: the preferable way was to decelerate in quartering rearward flight; a second way was to apply left pedal to control the divergence. When this second method was used, tail rotor power limits could be easily exceeded. When the left pedal became effective, recovery resulted in a rapid left yaw and the aircraft was recovered in forward flight. When yaw divergence was encountered with more rapid sideward velocity changes, the severity and rapidity of the maneuver was increased, making recovery by the second method less desirable and ultimately impossible at higher sideward accelerations within present tail rotor power limits.

79. Control margins during sideward flight were adequate; however, one control margin did not meet the requirements of the Army systems specification: the 0.7-inch aft cyclic control margin in 30 KTAS left sideward flight (corresponding to a 5.3-deg/sec pitch rate at 1.5 seconds). The lateral flight characteristics of the YAH-64 failed to meet the requirements of paragraph 10.3.9.1.1 of the systems specification, in that it was not always possible to stabilize at 30 KTAS in left sideward flight; and in that there was insufficient longitudinal control margin to produce a 15-deg/sec pitch rate in 1.5 seconds in translational flight to 35 KTAS in any direction relative to the nose of the aircraft.

Forward and Rearward Flight:

80. The variation of cyclic control position in low-speed forward and rearward flight was nonlinear. An abrupt change in gradient was observed at the airspeed for effective translational lift (about 20 KTAS), with the gradient change being more abrupt at the high-altitude test site. These longitudinal control gradient changes were not objectionable to the pilot. The variation of pitch attitude with forward airspeed was increasingly nose-up between 25 and 60 KCAS. This characteristic was objectionable during NOE flight, since it contributed to the restricted field of view due to pitch attitude, which is discussed in paragraphs 42, 72, and 90. The low-speed forward and rearward flight handling qualities of the YAH-64 are satisfactory but did not meet the requirements of paragraph 10.3.4.1.2 of the Army systems specification, in that the aircraft pitch attitude became more nose-up with increasing airspeed above 25 KTAS; and paragraph 10.3.9.1.1, in that there was insufficient longitudinal control margin to produce a 15-deg/sec pitch rate in 1.5 seconds in longitudinal flight to 35 KTAS in any direction relative to the nose of the aircraft. The noncompliance with paragraph 10.3.9.1.1 was previously discussed in paragraph 79.

Power Management

81. Power management of the YAH-64 was evaluated qualitatively throughout the engineering and operational tests. Throughout the aircraft envelope, no power management handling qualities problems were encountered. The most demanding power management maneuvers were the quick stops described in detail in paragraphs 143 through 145. During these maneuvers, rapid collective inputs were made from power for level flight at 120 KCAS to autorotation, and from minimum

power to topping power. The main rotor and engine speeds remained within limits without specific pilot compensation to maintain those limits (HQRS 2). Torque matching between the two engines throughout the engine torque range was excellent, and is an enhancing characteristic of the system. This characteristic is further discussed in paragraphs 143 through 145.

82. On several occasions the beep trim system was at least temporarily, and once completely, inoperative prior to the first flight of the day. An increase in rotor speed could not be accomplished on several occasions until a lengthy decrease was first commanded. The rotor speed beep trim system was often very slow to respond to an increase or decrease beep trim command. When the beep trim system functioned properly, it was smooth, accurate, and responded at a comfortable rate. The unreliable engine beep trim system is a shortcoming.

83. During engine start, the engine power levers must not be advanced from the ground-idle position to the fly position until engine oil pressure remains below 100 psi as the power levers are advanced. This restriction became particularly objectionable during engine starts in cool temperatures (between 5 and 15°C). A delay of between 1 and 5 minutes was necessary under these conditions before the power lever could be advanced to the fly position. This increased the time required to ready the aircraft for flight. At present, this time delay in advancing the engine power levers due to engine oil pressure limits is a shortcoming; however, during operation in much colder climates it is anticipated that delay times will increase, which could become a deficiency.

Mission Maneuvering Characteristics

84. Quantitative and qualitative evaluations of typical mission maneuvering characteristics were accomplished at 9500-, 4120-, 7000-, and 2303-foot test sites in desert and mountainous terrain. Aircraft agility and maneuverability were assessed during NOE, contour, and low-level flight; unmasking and masking maneuvers; operations from unimproved areas; dash and quick stops (accelerations and decelerations); lateral accelerations; and vertical displacements. The aircraft was evaluated against the mission maneuver criteria described in the Army systems specification.

Nap-of-the-Earth, Contour, and Low-Level Flights:

85. NOE, contour, and low-level flight maneuvers were conducted at 2303- and 7000-foot elevations over widely varying terrain at airspeeds ranging from zero to 120 KIAS. In the NOE combat environment, communicating effectively is of extreme importance. Pilot attention cannot be diverted from the flying task for the length of time required to tune the radios presently installed in the YAH-64. The lack of preset frequencies on the VHF and UHF radios is a deficiency. The problem of communication while flying in the NOE combat environment was greatly eased by the incorporation of an automatic radio transmitter selector system (ARTSS). The ARTSS, controlled by a button on the pilot cyclic stick, allowed

the pilot to individually select each of three radio transmitters without removing his hands from the controls. This is an enhancing feature which is particularly valuable while flying NOE.

86. Four shortcomings, three of which were previously discussed in this report, adversely affected NOE, contour, and low-level flight operations:

a. The pilot forward field of view was restricted by canopy distortion (further discussed in para 121) and by canopy structure. While flying NOE, the pilot exercised considerable compensation to adequately assure obstacle clearance (HQRS 5). The restricted field of view due to canopy structure and distortion is a shortcoming particularly evident in NOE flight.

b. The pitch-up from hover to 60 KCAS required an excessive forward cyclic input of approximately 3 inches to maintain acceleration (para 71).

c. Pitch-to-sideslip aerodynamic coupling increased pilot workload in maneuvering flight (paras 49 and 54).

d. Large longitudinal trim changes with power variations increases pilot workload when accomplishing maneuvers requiring significant power changes (*ie*, NOE in mountainous terrain, formation flying, etc) (para 43).

Unmasking and Masking Maneuvers:

87. Unmasking and masking maneuvers were conducted during the OT I testing phase at Edwards Air Force Base. The pilot was able to rapidly attain and precisely control hover altitude during masking and unmasking maneuvers. The inclusion of a radar altimeter as test instrumentation greatly aided the pilot in establishing and maintaining precise hover altitude, particularly in night OGE hover conditions during OT I tests. Although the hover characteristics of the YAH-64 were degraded by directional control characteristics (paras 80 through 84) in crosswinds of 10 KTAS or more, the hover stability of the aircraft was excellent under less critical wind conditions. The ability to rapidly attain and precisely control hover altitude is an enhancing characteristic. The unmasking and masking maneuver was satisfactorily accomplished. A radar altimeter should be installed in the production version of the YAH-64.

Operation from Unimproved Areas:

88. Operations from unimproved areas were conducted during the OT I tests at Edwards Air Force Base. The tail wheel configuration affords excellent unimproved area operational capability. Leveling the aircraft is not required prior to touchdown in these areas; the tail wheel configuration allows the aircraft to remain in a decelerating attitude prior to and during the touchdown. It also affords protection for inadvertent ground contact while flying NOE. The tail wheel configuration is an enhancing characteristic. Operations from unimproved areas are satisfactory.

Dash:

89. Rapid accelerations from zero airspeed to V_H were conducted during the OT I tests over a 3-kilometer level desert terrain course at Edwards Air Force Base. The power management system of the YAH-64 provided excellent automatic torque matching throughout the entire torque and airspeed range of the helicopter, an enhancing characteristic previously discussed in paragraph 81. The dash maneuver could be accomplished satisfactorily.

Quick Stop:

90. Rapid decelerations from high-speed flight to a hover at constant altitudes were qualitatively evaluated throughout the test program, and quantitatively evaluated at the 2303-foot test site. Quick stops were entered from airspeeds of 76 and 125 KTAS at an altitude of 50 to 75 feet above the ground. Power was rapidly reduced and the aircraft was flared to decelerate at a constant altitude. During quick stops from 125 KTAS the aircraft was actually in autorotation. Recovery from the quick stop was accomplished both to an OGE hover and to a topping power vertical climb. Quick stop maneuvers were satisfactory and are further discussed in paragraphs 143 through 145. The excellent automatic torque matching (para 81) and tail wheel design (para 88) are enhancing characteristics previously discussed. The restricted forward field of view due to nose-high attitude (paras 42 and 72) was a shortcoming particularly noticeable during quick stops.

Lateral Acceleration

91. Right lateral accelerations were conducted at the 4120-foot test site under the conditions listed in table 2. Left lateral accelerations were prohibited by the SOFR (ref 4) due to the loss of directional control in 25 to 30 KTAS left sideward flight, as discussed in paragraph 77. Time histories of right lateral acceleration are presented in figures 93 through 95, appendix G. The maneuver was initiated from a vertical climb by applying right lateral cyclic to establish the desired roll attitude. Collective control was used to maintain a constant altitude. An incremental build-up in roll attitude was accomplished to a maximum roll attitude of 14 degrees. Roll attitude was maintained as the aircraft accelerated. Aircraft response to lateral cyclic control was quick and positive, allowing rapid attainment and good control of roll attitude during the maneuver (HQRS 3). As the aircraft accelerated to the right, increasing left directional control was applied to maintain heading until the tail rotor gearbox power limit was reached, at which point the aircraft was allowed to yaw out of the maneuver into forward flight. The yaw recovery was initiated between 28 and 32 knots after traversing 100 to 250 feet horizontal distance. No handling qualities problems were encountered. The limiting factor in the maneuver was tail rotor gearbox power limitations. Directional control limits were not reached.

Vertical Displacement

92. The vertical displacement maneuver was evaluated at the 4120-foot test site under the conditions listed in table 1. Representative time histories of this maneuver

are presented in figures 96 and 97, appendix G. The maneuver was initiated from a level flight airspeed of 140 KTAS with an aft cyclic control step input. Collective was held fixed during the maneuver. A build-up in normal acceleration was conducted to a normal acceleration of 1.8g. Immediately following application of aft cyclic control, a normal acceleration spike was felt by the pilot. Inspection of the data revealed only the normal g spike due to the control input, followed by a concave downward shape of the normal acceleration trace which is characteristic of positive maneuvering stability. A slight roll-to-pitch coupling (right roll with nose up) is shown in the data, but was not observed by the pilot. Handling qualities during the vertical displacement maneuver were satisfactory.

Weapons Firing

93. The aircraft was evaluated under the test conditions listed in table 2 for the following general characteristics: vibration/noise levels; airframe reactions; gun barrel excursions; weapon gas in the cockpit; FOD during case ejection/rocket mortar debris; and flash effects during night firings. Weapons fired were the Hughes 30mm single-barrel chain gun mounted below the copilot, and wing pod-mounted 2.75-inch FFAR. The 30mm ammunition was inert rounds without tracers, and the 2.75-inch FFAR had inert 10-pound warheads. Total ammunition fired was 510 30mm rounds and 106 2.75-inch FFAR. Gun and rocket master armament switches were controlled by the pilot or the copilot/gunner. The gun/rocket firing circuits were armed by selecting either guns or rockets on the armament panel (fig. 36, app B). The number of 30mm rounds to be fired and gun barrel positions could only be set from the gunner position for DT I testing. Once the systems were armed the pilot or gunner was required to activate the weapons action switch located on the cyclic control (fig. 5, app C) with his thumb and commence firing with the cyclic trigger. During firing, the pilot or copilot/gunner could not communicate or retrim the aircraft. The undesirable location of the weapons action switch is a shortcoming. Aircraft response to 30mm gun firings is presented in table 9.

94. Most aircraft responses required only minimal pilot compensation to readjust the projectile impact point. Although the gun barrel exhibited minor circular excursions, it did not seem to affect accuracy, since ground impact revealed small dispersions, especially in forward flight. Ejected 30mm casings did not contact the aircraft under any firing condition. Vibration levels during firing were low and in the medium to low frequency range. Gun gases entered both cockpits regardless of gun position and flight condition. Although the gas concentration was low, it became annoying to both pilots after several firing bursts. Additionally, these gas odors could mask other significant odors such as hydraulic/electrical or fuel fumes. Weapons gas fumes in the cockput during weapons firing are a shortcoming.

95. Aircraft reaction to 2.75-inch FFAR firings was minimal. Gas fumes expelled from the rockets entered both cockpits, creating an annoying odor, especially after ripple firing. The presence of weapon gases in the cockpit during firing is a shortcoming which was discussed above.

Table 9. Aircraft Response to 30mm Gun Firing.

Gun Barrel Position	Burst Length (rounds)	Flight Mode	Aircraft Reaction	Handling Qualities Rating Scale
Forward, 60 deg down	30	Hover	Slight initial pitch-up tendency	2
90 deg right (left), 60 deg down	30	Hover	Slight left (right) yaw	3
90 deg right (left), zero deg down	30	Fwd flight, 60 and 90 KIAS	Slight left (right) yaw	2
90 deg right (left), 60 deg down	30	Fwd flight, 60 and 90 KIAS	Slight initial left/right yaw and pitch-up, and mild left (right) rolling tendencies	3

96. Night firing of the 30mm gun and the 2.75-inch FFAR revealed very low flash levels for the 30mm gun and normal flash levels for the 2.75-inch FFAR. Rockets fired in the ripple mode caused momentary loss of forward field of view due to flash blinding, but posed no significant problems in handling the aircraft. The flash levels of the 30mm gun and the 2.75-inch FFAR during night firing are not excessive.

Instrument Flight

97. Instrument flight characteristics were evaluated throughout the test. However, one specific flight was conducted at night to best simulate IMC. Navigational equipment installed limited the evaluation to basic instrument pilot tasks, *ie*, climbs and descents, standard rate turns, level flight, airspeed changes, ITO's, and simulated ground controlled approaches (GCA's).

98. The pilot flight instruments were grouped in a standard basic instrument configuration; however, the size of the present vertical situation display (VSD) made the instrument separation wider than a normal grouping. This resulted in increased pilot workload during instrument scan. The wide dispersion of the basic flight instruments is a shortcoming.

99. The ITO procedure was to apply climb power from a stabilized 5-foot wheel height hover (IGE), then rotate to a 4-degree nose-up pitch attitude, and accelerate to the best rate of climb airspeed (75 KIAS). Between hover and 65 KIAS, aircraft pitch attitude changed 8 degrees nose-up, requiring an excessive forward cyclic input (3 inches). The large trim change required during acceleration decreased flight path accuracy and increased pilot workload (HQRS 4). This shortcoming was previously discussed in paragraph 71. Except for the ITO, basic instrument pilot tasks were easily accomplished (HQRS 2) due to the stable flying qualities of the aircraft with SAS ON. Airspeed and altitude could easily be maintained within ±5 knots and ±50 feet, respectively. The essentially deadbeat short period and excellent gust response characteristics of the SAS also contributed to the satisfactory instrument flying characteristics under turbulent flight conditions. SAS OFF flight resulted in increased pilot workloads to hold within airspeed and attitude constraints (HQRS 3); however, no observed handling qualities characteristics should limit the envelope of the aircraft following SAS failure. The excellent SAS OFF handling qualities are an enhancing characteristic of the YAH-64 which is further discussed in paragraph 101. Flight path stability was evaluated by flying simulated GCA's using the runway visual approach slope indicator (VASI) system at various airspeeds (70 to 110 KIAS). Rate of descent and airspeed were easily controlled and small corrections to heading were easily accomplished (HQRS 2). Due to the requirement for continual retrimming during GCA's, the simultaneous release of trim holding forces in all axes after depression of the trim release button became an annoying characteristic. The lack of single-axis trim capability is a shortcoming. The present avionics configuration in the YAH-64 requires the pilot to switch hands on the controls and direct his line of sight down and to the right to change radio frequencies. Since the present UHF and VHF radios require an extensive amount of time to retune due to lack of preset frequencies, the pilot

must spend an excessive amount of time with his attention diverted from the primary task of instrument flying. The lack of preset frequencies on the UHF and VHF radios is a deficiency previously mentioned in paragraph 85.

Systems Failures

Simulated Stability Augmentation System Failures:

100. Aircraft response to SAS hardover failures was evaluated at the conditions shown in table 2. SAS hardovers were electrically introduced through a test hardover box. A part of the ASE system was the SAS channel hardover monitor (para 12, app C), which functioned to disable individual SAS channels within 0.2 second following a hardover. Representative time histories of SAS hardovers are presented in figures 98 through 101, appendix G. Pilot warning of a hardover was illumination of a master caution light and an angular acceleration. Little or no pilot response was required (HQRS 2). The SAS hardover characteristics exhibited by the aircraft are satisfactory. The SAS monitor system is an enhancing feature.

101. Simulated SAS failures (disengagements) were qualitatively evaluated throughout the flight envelope. Initial pilot response to a simulated SAS failure was minimal, as the aircraft flight path remained essentially unchanged. The aircraft did exhibit an aperiodic long-term response, SAS OFF, but of a relatively high time constant which did not interfere with normal pilot tasks. Basic maneuvers were performed SAS OFF with minimal pilot effort. Although the handling qualities of the aircraft are degraded with the SAS OFF as compared to SAS ON, the excellent SAS OFF handling qualities are an enhancing characteristic (HQRS 2).

Simulated Electrical System Failures:

102. The failure mode evaluation of the electrical system consisted of turning off either or both generators and the battery. Failing the No. 1 generator resulted in the No. 2 generator picking up all the load of generator No. 1 and illumination of a caution capsule light. Failing the No. 2 generator resulted in a mild SAS input to the rotor system due to a surge in the electrical power supply. Failure of both generators resulted in the loss of SAS, VSD, Marconi gauges, and cockpit panel lighting. The minimal effect of the failure of a single generator is an enhancing characteristic.

Single-Engine Failures and Autorotational Entries:

103. Aircraft response characteristics following a simulated single-engine failure from dual- and single-engine flight were evaluated at the test conditions listed in table 2. Simulation of the engine failure was accomplished by a rapid power control lever (PCL) movement to the flight-idle position. The controls were held fixed for 2 seconds or until transient rotor speed limits were reached. Representative time histories are presented in figures 102 through 104, appendix G. The aircraft

engine-out warning system consists of a modulating (1 Hz) aural tone and engine-out/rotor speed low warning light in each cockpit, activated by the parameters presented in table 10.

Table 10. Aircraft Engine-Out Warning Systems.[1]

System	Activated By	Pilot Presentation
Engine	$N_P \leqq 96\%$ or $N_G \leqq 67\%$	Aural tone and engine-out light
Rotor	$N_R \leqq 90.3\%$	Aural tone and rotor speed low

[1]With PCL in flight-idle position, N_P system is deactivated.

104. A time history of aircraft response following a simulated single-engine failure at 79 KCAS level flight is presented in figure 102, appendix G. Rotor speed decay (3 percent per second) began essentially at throttle chop and rotor speed stabilized at 98.6 percent (285 rpm) without collective movement. Aircraft response was negligible, exhibiting little or no tendency to vary from level flight attitude and airspeed. Recovery required very minimal pilot compensation (HQRS 2). The primary pilot cue came from a slight left yaw (2 degrees). The No. 1 engine-out warning light was not activated due to gas generator speed (N_G) stabilizing slightly higher than 67 percent.

105. A time history of a simulated single-engine failure during a 2000-ft/min rate of climb at 76 KCAS is presented in figure 103, appendix G. A rotor speed decay of 5 percent per second began 0.5 second after throttle chop, resulting in a transient droop to 91 percent rotor speed (N_R). After collective movement was delayed for 2 seconds, the rotor speed decay was quickly arrested. The aircraft exhibited a slight left yaw rate of 3 deg/sec and a slight left roll rate of 2 deg/sec. Pilot response during recovery required minimal effort (HQRS 3). The primary pilot cues were a noticeable left yaw accompanied by the low rotor speed aural warning.

106. A time history of a simulated No. 1 engine failure from single-engine level flight (autorotational entry) at 79 KCAS is presented in figure 104, appendix G. Rotor speed decay of 5 percent per second began 0.5 second after initial throttle chop with a minimum transient rotor speed of 80 percent N_R. Collective movement was initiated after 2 seconds (85 percent N_R), which immediately retarded rotor speed decay. Aircraft response was mild, exhibiting rates of 5 deg/sec left yaw, 5 deg/sec pitch-down, and a ±2-deg/sec pilot-induced roll oscillation. Primary pilot cues were the left yaw, an audible rotor speed decay, and a low rotor speed aural warning. Pilot responses required to initiate

autorotational entry were minimal (HQRS 3). Immediately following a rapid reduction of the collective to enter autorotation, minimal cyclic and directional control inputs were required to reach a steady-state condition. The aircraft exhibited no rotor overspeed tendency. Rotor speed control during left and right maneuvering turns was excellent, resulting in a considerable reduction in pilot workload during a normally high workload flight condition. The excellent rotor speed control in autorotational descent is an enhancing characteristic. The aircraft response to single- and dual-engine failure from a single-engine condition is satisfactory.

STRUCTURAL DYNAMICS

Vibration Characteristics

107. Vibration characteristics of the YAH-64 were qualitatively evaluated throughout the test program and quantitatively evaluated at the conditions listed in table 2. Figures 105 through 117, appendix G, depict the pilot vibration environment. Seat vibrations were generally low vertically and laterally and near zero longitudinally. The 4/rev vibrations increased as airspeed was decreased in the approach to landing; however, the levels were not objectionable to the crew, due to the short time the aircraft was operated in this mode. Lateral and vertical vibrations of the cyclic stick were unnoticeable to the pilot. Instrument and panel vibrations did not affect the readability of the instruments.

108. Figures 118 through 123, appendix G, depict the copilot/gunner vibration environment. Vibrations at this station were qualitatively evaluated to be higher than those at the pilot station.

109. Figures 124 through 136, appendix G, present vibration data from numerous sensors throughout the aircraft. Center of gravity vibrations were primarily 4/rev and peak measured amplitude was 0.1g in the vertical direction. Left and right avionics bays exhibited different characteristics. The left bay showed significant vibrations in the 1/rev vertical and longitudinal axes, while the right bay vibrations were primarily 4/rev. Left wing 4/rev vibrations reached 0.92g in the vertical axis. These vibrations were not felt in the crew station, nor were they visually observed.

110. Selected vibration parameters measured on aircraft SN 74-22249 are presented in figures 137 through 150, appendix G. These plots are presented to allow a comparison of vibration characteristics between the two prototype aircraft and to indicate the effects of cg variations. Aircraft SN 74-22249 was qualitatively judged to have lower vibration levels than aircraft SN 74-22248. No significant cg vibrations were noted. Figures 151 through 153 present the effect of maneuvering flight on vibrations. The pilot observed a general increase in vibrations primarily at the 4/rev frequency as normal load factor increased.

111. 30mm gun firing did not alter the crew compartment vibration characteristics, other than to introduce a vertical vibration at the copilot station

at the gun firing frequency. These vibrations were not annoying. Data recorded during 30mm weapons firing indicated that avionics bay vibrations may be high during firing; however, firing bursts were not of sufficient duration to allow analysis of those vibrations. Launching 2.75-inch FFAR's had no noticeable effect on aircraft vibration levels.

112. Vibration levels were assessed as very low at operational airspeeds up to 120 KCAS. This qualitative assessment was probably due to a combination of two factors: the generally low level of the physiologically significant vibrations, and the high frequency of all vibrations of significant magnitude. Studies have shown that the human body is most sensitive to vibrations in the 2- to 12-Hz range (torso), and the 20- to 30-Hz range (head and eyes) (ref 10, app A). The YAH-64 1/rev (4.8 Hz) and 2/rev (9.6 Hz) vibrations were consistently near zero. Vibrations in the 4/rev (19.3 Hz) range were apparently absorbed by the torso, because they were not felt at the head. Vibration levels felt by the crew increased at high speed in the 8/rev (38.5 Hz) lateral direction at the copilot seat but due to the high frequency, this increase was not objectionable. No noticeable blurring of instruments was reported by either crewmember during any of the aircraft maneuvers accomplished during this evaluation. Another contributing factor to the overall crew evaluation of low vibration levels was the absence of longitudinal vibration. The low level of physiologically significant vibrations at the crew stations is an enhancing characteristic.

Control System Loads

113. During contractor developmental testing, high structural control loads were observed in the collective and lateral cyclic actuators. Consequently, the flight envelope of the YAH-64 was limited by the SOFR (ref 4, app A), and collective and lateral cyclic actuator load indicators were installed at the pilot station. These control system structural load indicators had to be monitored during maneuvering flight to assure that control system load limits were not exceeded.

HUMAN FACTORS

Aircraft Preflight Inspection

114. The preflight inspection was evaluated for sequence and items requiring checking throughout the test program. The sequence of preflight/start items as outlined in the pilot's pocket checklist was satisfactory; however, 13 shortcomings were noted during the preflight inspection procedure:

a. Lack of quick access to the interior of the forward avionics bays. This will be required when using encoded transponder responses and voice security equipment.

b. The piano-hinge pins on the catwalk doors are not positively secured in the hinge.

c. The bolts securing the rotor blade strap packs to the rotor head are mounted with the nuts on top. The bolt could fall out if a nut came loose or shattered due to battle damage.

d. Lack of sufficient lighting on the APU and engine oil sight gauges.

e. The APU oil sight gauge is too sensitive to aircraft attitude. Accurate reading occasionally requires the aircraft to be rocked.

f. Transmission compartment access doors cannot be latched open, making preflight inspection in this area awkward.

g. The utility hydraulic reservoir sight gauge is obscured by the backup control system detent spring on the lateral control actuator.

h. The fluid level markings on the utility hydraulic reservoir are decals and are subject to deterioration.

i. The difficulty in reading the primary hydraulic reservoir fluid level. The fluid level markings are on the opposite side of the metal scale from the fluid level pointer.

j. Loss of continuous electrical power when transitioning from aircraft power to ground power puts unnecessary power surges on the avionics equipment.

k. The inability to check the main transmission oil sight gauges without removing the side panels (12 fasteners) or using an inspection mirror.

l. Lack of proper steps above the wing makes inspection of the rotor head difficult and requires the pilot to utilize areas on the engine fairings not specifically designated as step areas.

m. The forward engine door fasteners can be inadvertently unlatched by stepping on them while inspecting the rotor head.

<u>Cockpit Evaluation</u>

115. The pilot and copilot/gunner cockpits were qualitatively evaluated throughout the test program for ease of entry and egress, pilot/seat interface with controls, instrument arrangement and readability, field of view, and crew comfort. Four enhancing characteristics, one deficiency, and 21 shortcomings were noted in addition to the other characteristics pertaining to the cockpit which are discussed elsewhere in this report where they impact directly on the task under discussion.

116. The four enhancing characteristics are as follows:

a. The caution light enunciator panel system flashed only the most recent caution light, although several other caution lights may have previously been

illuminated. The system was reset by pressing the flashing master caution light. The flashing caution panel effectively emphasizes systems problems.

b. The pilot left overhead circuit breaker panel provided quick identification and access to circuit breakers. Additionally, the long-stem circuit breakers will aid in pulling the circuit breakers should the need arise.

c. The Marconi digital fiberoptic instrument presentation allowed rapid interpretation of parameters, and the small multigauge panel located in the forward cockpit provided six individually selectable digital readout parameters.

d. The incorporation of a rotor brake significantly reduced blade flapping during shutdown in gusting wind conditions.

117. The lack of preset frequencies on the VHF and UHF radios required excessive hands-off control time and redirection of pilot attention to change radio frequencies. This was particularly noted during NOE maneuvers and simulated IMC flight, and is a deficiency which was discussed in paragraphs 85 and 99.

118. The 21 shortcomings noted are as follows:

a. Lack of foreign object protection boots where directional controls pass through the floor.

b. The present configuration of the cockpits restricts the passing of essential maps or other mission-related information between the pilot and copilot.

c. The position of the power levers in the OFF position blocks pilot and copilot view of certain fuel management switches. This increases the likelihood of improper fuel switch positions during a start.

d. The canopy drumming in the aft cockpit increases pilot fatigue. This was more noticeable on aircraft 74-22248 than on aircraft 74-22249.

e. The copilot canopy door handle cannot be reached by a 95th percentile pilot when secured in his seat. This requires a crewman to close the canopy door.

f. The inability to read the lighted caution panel segments in bright sunlight. This requires hands-off control time to shade the panel for positive identification of a certain light should the master caution light illuminate.

g. The pilot and copilot directional control pedal adjustment knobs are poorly located and require an excessive number of turns to adjust pedal positions.

h. Lack of pilot and copilot longitudinal seat adjustment.

i. Pilot and copilot cockpit entrance and egress is hindered by the cyclic stick.

j. Seat vertical adjustments are difficult to operate. Positive seat locking is difficult to detect, which could possibly result in takeoff with the seat not fully secured.

k. The lack of floor ramps (heel rests) under the the copilot directional control pedals.

l. Lack of a magnetic compass in the forward cockpit.

m. The common gyro for pilot VSD and turn needle. Loss of that gyro results in loss of the pilot's two primary turn reference instruments.

n. The copilot floor radio/intercom switch is awkward to use due to its mounting angle and location.

o. Poor mounting of the pilot instrument panel ECS vents.

p. The APU start switch can be inadvertently activated when climbing into or out of the pilot station.

q. Lack of adequate secure storage space in the cockpit.

r. The rotor brake can be inadvertently locked with the switch guard in place, causing rapid and possibly damaging rotor stoppage.

s. After prolonged ground operation with the engines shut down and with electrical power applied to the aircraft by ground power units or the APU (*ie*, a large number of caution lights illuminated) the caution panel becomes extremely hot. Items coming in contract with this panel may be damaged.

t. Lack of a turn needle in the forward (copilot/gunner) cockpit.

u. Cockpit entrance step mountings are too weak. During pilot training, one step mounting broke, causing minor injury to an aircraft crewman.

The rotor brake actuator switch failed to meet the requirements of paragraph 3.7.1.4 of the Army systems specification, in that the rotor brake could be inadvertently placed in the lock position with the switch guard in place.

119. Although a number of shortcomings have been noted, the overall cockpit is a significant improvement in attack helicopter cockpit design. Within the temperature range encountered during this test (50 to 110°F), the ECS was very effective. The ability to run the ECS with only the APU operating will serve to greatly reduce pilot fatigue.

Night Lighting Evaluation

120. The evaluation of night lighting characteristics was conducted during several OT I tests and one DT I flight to evaluate night lighting and interior/exterior reflection characteristics. Flight profiles were conducted over sparsely populated desert areas and low population density towns. Visual landing approaches were made to both lighted and unlighted landing sites.

121. The visual distortion in the canopy (fig. C), primarily in the windshield/copilot overhead interface area and in the copilot side canopies (when viewed from the pilot's position), presented more of a problem at night than during the day. In daylight the pilot can take his visual cues from peripheral vision or direct his line of sight to the side through relatively distortion-free pilot side panels. At night, however, the peripheral vision cues came from the previously mentioned areas where distortion is a problem. The distortion of runway threshold and boundary lights is particularly annoying during night landings. The forward-aimed landing light also tends to direct pilot attention forward through the area of maximum canopy distortion. The mild disorientation produced by canopy distortion could be alleviated by focusing the searchlight 30 to 45 degrees either side of the aircraft nose, so that the pilot could take his visual cues through the relatively distortion-free canopies that form the side of the pilot station. While this solution is adequate for hovering and low-speed forward flight, over improved areas the visual distortion along the aircraft's flight path during night NOE or contour flying will seriously degrade the capability of the YAH-64 to perform these missions. The distorted pilot field of view through the copilot overhead and side canopies is part of a shortcoming which was previously identified in paragraph 86.

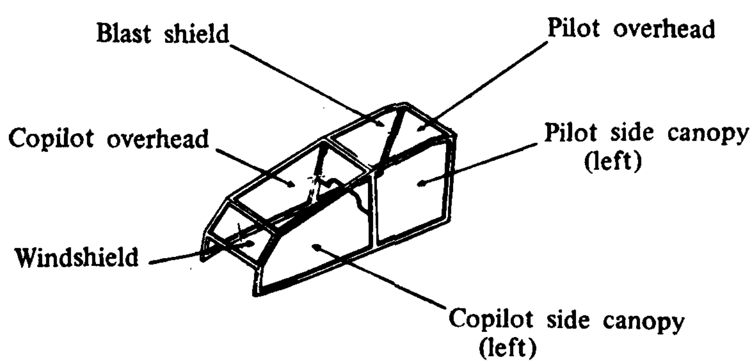


Figure C. Canopy Nomenclature.

122. Interior light reflections from both exterior and interior light sources were generally low and posed no obstruction to the pilot's field of view. Canopy reflections of interior instrument lights viewed by the pilot were localized to the copilot right side canopy, the left forward portions of the copilot overhead panel (copilot instrument lights), and a segment of canopy curvature on the forward portion of the pilot side canopies (pilot instrument lights). From the copilot station, reflections were localized to a segment of the forward curvature of the copilot side canopies (copilot instruments). The intensity of all these reflections could be lowered to an acceptable level by use of the instrument light dimming rheostats. After suitable adjustment of instrument light levels, these reflections posed no restriction to the pilot or copilot field of view. With instrument lights on, the master caution light was too dim to adequately draw the pilot's attention to the caution light panel. Excessive dimming of the master caution light is a shortcoming.

123. The most annoying reflection due to interior lighting was caused by the pilot map light located on the pilot left upper canopy rail adjacent to the circuit breaker panel. Operation of this light produced an objectionable reflection in the blast shield. The reflection of the pilot map light by the blast shield is a shortcoming.

124. Interior canopy reflections due to an exterior light source (*ie*, aircraft strobe lights, other aircraft, and ground light) were minimal. Some low-intensity reflection of ground light was observed by the pilot in the copilot overhead panel. These reflections extended about 2 inches inboard from the canopy frame in the aft two-thirds of the panel. There was no interference with the pilot field of view through the copilot overhead panel and the reflections were only noticeable when flying over brightly lighted populated areas.

125. Primary cockpit lighting was satisfactory in both coverage and intensity of illumination. A reflection of interior lights was observed in the face of the VSD but was not objectionable to the pilot. No other gauges exhibited any reflections. The VSD has a wide range of light intensity adjustment suitable for all lighting conditions from bright daylight to minimum-light night flight. For night flight a red filter is included as a separate item and thus is subject to being lost or damaged.

126. Auxiliary floodlights are provided for cockpit illumination and backup instrument lighting. The pilot may select red or white floodlights but no intensity control is available. This lack of light intensity control for the red and white flood lights is a shortcoming. The area coverage of the red floodlights does not adequately cover the aft one-half of either side console and is a shortcoming. Items not adequately illuminated by the red floodlights include the fuel panel, lighting control switches and rheostats, anti-ice panel, VHF and UHF radios, cabin air control panel, and APU control panel. Of these items only the radio illumination would be critical when using the floodlights with failure of the primary lights. The map light is capable of providing emergency lighting of the left side console but not the right console.

127. The circuit breaker panels are provided with red backlighting. The circuit breaker panel on the upper left canopy rail is easily read at night; however, the lower left circuit breaker panel cannot be adequately seen under either day or night conditions, which is a shortcoming. A tripped circuit breaker is identified by a white band around the base of the circuit breaker stem. At night this white band is not sufficiently visible to allow rapid identification of a tripped circuit breaker. The inability to identify a tripped circuit breaker on either panel at night is a shortcoming. The backlighting of the canopy rail circuit breaker panel should be controlled by an ON/OFF switch in series with the intensity-controlling rheostat. This would allow the pilot an option to turn the circuit breaker panel lights off to reduce the likelihood of aircraft detection by the enemy at night.

128. The operation and effective light patterns of the landing and searchlights were evaluated during landings to lighted and unlighted areas and during ground taxi operations. Both lights provided satisfactory intensity and area coverage. The searchlight intensity was less than the landing light, but provided excellent illumination of the area on either side of the aircraft. Searchlight slew rate was satisfactory. Control of both the landing light and the searchlight was unrestricted and allowed positive tactical operation with minimal pilot distraction.

RELIABILITY AND MAINTAINABILITY

Auxiliary Power Unit

129. A T62T-40-4 APU, manufactured by the Solar Division of International Harvester Company, was installed. A 3000-psi utility hydraulic accumulator provided power for starting. The starting sequence was automatic, relying on a self-monitoring system to abort the start if any one of the following five parameters exceeded their limits: low oil pressure (6 psi at 90 percent N_G; high EGT (1350°F); N_G overspeed (110 percent); N_G backdown (engine deceleration from 90 to 75 percent N_G), and N_G underspeed (not achieving 70 percent N_G within 1 minute).

130. When the APU accelerated to 80 percent N_G, the APU drive clutch engaged, causing the APU to drive the accessory gearbox, and 30 seconds after initiating the start, the shaft-driven compressor throttling valve opened causing an SDC load on the accessory gear box. During a normal start, the APU reached 100 percent N_G in approximately 12 seconds. The APU frequently failed to accelerate to 100 percent within 30 seconds, at which time the shaft-driven compressor throttling valve would open, causing N_G to decrease to 75 percent, thus causing the starting sequence to be aborted. Prior to attempting a restart, the accumulator had to be recharged by a hand pump, which required up to 5 minutes. The excessive utility hydraulic accumulator manual recharging time is a shortcoming. Additional factors resulting in aborted starts were weak ignition and the fuel control acceleration schedule, which was too sensitive to small changes in density altitude. The unreliable APU starting system is a deficiency.

Pneumatic Starter

131. The YT700-GE-700 engine incorporates a pneumatic starter designed to motor N_G to 40 percent. Primary compressed air for the starters comes from the shaft-driven compressor, utilizing an air start cart and/or the bleed air from the left engine as secondary sources. The normal starting sequence involved directing air from the shaft-driven compressor to the starter until 20 percent N_G was achieved (approximately 7 seconds) and then moving the PCL to ground-idle. A ground-idle of 67 percent N_G would normally be reached within 25 seconds. However, during an estimated 15 percent of engine starts attempted, 20 percent N_G could not initially be achieved. The procedure then was to turn off the ECS, allowing a larger volume of air to start the engines. Primary causes of the low N_G on starts were leaking integrated pressure air system (IPAS) manifolds, inoperative air surge valve, partially open starter valve, and leaking air lines. The continual recurrence of engine start problems will significantly affect the operational readiness of the AAH in the field. The unreliable air start system for the engines is a deficiency.

Fan and Hanger Bearing Assembly

132. The IR suppression system of the engines used high-volume cooling air supplied by a tail rotor drive shaft mounted impeller (fan). The fan was mounted aft of the transmission accessory gearbox as an integral part of the No. 1 tail rotor drive shaft. Fan balance was critical, due to its size and weight. The fan has two critical frequencies of 62 percent (3002 rpm) and 85 percent (4164 rpm). Fan imbalances generated high structural loads at the No. 1 tail rotor drive shaft hanger bearing assembly when operating at or near the critical frequency. Because of the ultimate consequences of the fan or hanger bearing assembly failure (*ie*, loss of directional control), the vibration loads were monitored continuously, using a light in the cockpit which illuminated when hanger bearing loads reached 250 pounds. Several times during DT I fan vibrations reached a level of concern requiring removal and rebalancing. On one occasion, during a maintenance run-up, the fan failed at the shaft neck, causing minor internal damage and failure of the tail rotor drive shaft. Contributing factors toward fan unbalancing were foreign object damage and an accumulation of oil/hydraulic fluids which allowed accumulation of dust. The structurally inadequate tail rotor drive shaft-mounted cooling fan is a deficiency.

Miscellaneous Failures

133. The caution light system for warning of low oil pressure in the main and accessory gearboxes receives inputs from pressure-sensing units mounted at critical positions on both boxes. During DT I testing, numerous flights were aborted due to illumination of a low oil pressure warning light for the main or accessory gearbox. Maneuvers involving high side forces or normal accelerations less than approximately 0.2g caused these lights to illuminate. Most postflight inspections revealed a faulty pressure switch. The unreliable low oil pressure switches for the main and accessory gearboxes are a shortcoming.

134. The aircraft employs two fuel cells: 151 gallons forward and 202 gallons aft (para 18, app B) for a total of 353 gallons or 2294 pounds of JP-4 fuel. On several occasions, fuel was siphoned from the forward tank to the rear tank at a slow rate. Although this was not significant during flight (and could be eliminated by transferring fuel), long periods of hangar time resulted in significant quantities of fuel being transferred which affected the aircraft cg. The siphoning of fuel from the forward tank to the aft tank is a shortcoming.

135. The fuel quantity gauges, manufactured by Canadian Marconi (installed on aircraft SN 74-22249), displayed fuel state to within a 10-pound accuracy but zero shifts causing major errors in the indication of total fuel remaining were experienced. With the aircraft on the ground a flameout of the left engine and APU occurred due to fuel exhaustion in the forward fuel cell. At the time of the flameout, the Marconi fuel gauge indicated 160 pounds of usable fuel. The zero shift of the Marconi fuel quantity gauges is a deficiency. The fuel low warning system functioned properly.

136. The Marconi gauges occasionally went to half-gain (one-half of the light segments extinguished) on the No. 1 engine parameters, and the No. 2 engine parameter lights were extinguished. This occasional gauge system failure required the simultaneous pulling and resetting of the instrument primary circuit breaker in both cockpits to restore the gauges to normal. This was particularly annoying during night or simulated IMC flight conditions. The intermittent Marconi gauge failure is a shortcoming.

137. An excessive amount of oil is thrown out by the engines. This increases maintenance problems by trapping dirt and sand from unimproved area operation, as well as requiring extra crew workload for airframe cleaning.

138. The lead-acid battery venting outlet is located close to the external power receptacle. Fumes can come in contact with an electric arc or could cause corrosion of the external power receptacle.

SUBSYSTEMS TESTS

Engine Performance Characteristics

139. During performance testing engine performance and inlet and exhaust data were obtained. The performance helicopter was equipped with calibrated engines which were not changed during DT I testing. The inlet and exhaust data were used to correct the engine computer deck (73004) to obtain engine power available and fuel flow for the YAH-64 installation. The performance requirements of the Army systems specification were checked for compliance using the engine specification data corrected for installation loss and power-required data obtained from the performance tests. The uninstalled YT700-GE-700 engine is rated at 1536 shp at the intermediate (30 minute) limit (IRP) and 1250 shp for MCP under zero airspeed, sea-level, standard-day conditions. As installed in the YAH-64, a

specification engine mounted on the left side would produce 1517 shp (IRP) and 1232 shp (MCP) at sea-level, standard-day static conditions for installation losses of 19 and 18 shp, respectively. The right engine losses were slightly higher and amounted to 23 shp at IRP and 21 shp at MCP under sea-level, standard-day static conditions. The power-available data, as corrected for installation losses, are presented in figures 154 through 162, appendix G. Power losses (power not available to the main rotor and tail rotor) associated with the accessories and drive train were determined by comparing the total engine shp to the summation of main and tail rotor shp and are presented in figure 163. The power loss determined by this comparison ranged from 121 to 175 shp over the conditions tested.

140. The engine inlet and exhaust characteristics determined during the GCT are presented in figures 164 through 173, appendix G. The inlet temperature rise and inlet pressure over ambient conditions are presented for various flight conditions. Figures 164 and 165 show the inlet characteristics in forward and rearward flight. At airspeeds below 20 KTAS, the inlet temperature and pressure show a slight rise, reaching a maximum temperature rise of 1 degree Celsius and a total to static pressure ratio of 1.0036 at OGE hover conditions on the right engine. Above 20 KTAS, the inlet temperature and pressure data essentially followed 100 percent ram recovery, but were displaced by 0.4 degree for the left engine and 0.5 degree for the right engine. The inlet characteristics for sideward flight are shown in figures 166 and 167 and the variation with sideslip at various forward airspeeds is shown in figures 168 and 169. Figures 170 and 171 show that the inlet characteristics did not vary significantly with vertical speed. The engine exhaust data are shown in figures 172 and 173, and are presented in terms of the parameters required by the GE engine deck. The exhaust duct static pressure rise coefficient is presented as a function of the power turbine swirl angle. These terms are defined in appendix F.

141. The engines installed in the YAH-64 performance test vehicle had different fuel control models. Engine SN 207258 (right engine) had a Group 10 hydromechanical unit (HMU) and engine SN 207261 (left engine) had a Group 11 HMU. In general, the right engine burned more fuel, ran at a lower gas temperature, and developed more power than the left engine. The static engine performance obtained from level flight and hover testing is presented in terms of referred parameters in figures 174 through 198, appendix G. Data from the test cell calibrations and the engine specification are also presented. Generally, the installed engines produced slightly less power and used more fuel than during the test cell calibration.

142. The engine design is such that the maximum power available is limited by four different parameters. These parameters are measured gas temperature ($T_{4.5}$), gas generator speed (N_G), referred N_G, and fuel flow. These parameters limit power as a function of engine inlet temperature and pressure, as shown on figure 199, appendix G. At sea level these limits are approximately as follows: N_G above 35°C, $T_{4.5}$ at 8.5 to 35°C, fuel flow at 8.5 to -9°C, and referred N_G below -9°C (fig. 199). The fuel flow limit is present up to an altitude of approximately 750 feet. At this point, the change from the $T_{4.5}$ and referred N_G limits occurs

at approximately 1°C. The temperatures where these limits change generally decrease slightly with increasing altitude. Additionally, the temperatures where these limits change can be significantly different for various engines. This is especially true for the change between the $T_{4.5}$ and N_G limits which, for some engines, appear to be as low as 15°C. The significance of this to the pilot is that when operating at moderate to high ambient temperatures, one engine may be at the $T_{4.5}$ limit and the other engine unable to reach the $T_{4.5}$ limit due to N_G limiting in the engine control unit (ECU). Further, when operating at colder ambient temperatures (0°C and below), the engine may develop considerably less power than would be expected. Figure 196 presents the estimated and installed operating schedule for the YT700-GE-700 engine.

143. Engine dynamic response characteristics were qualitatively evaluated throughout this evaluation and specific quick-stop maneuvers were accomplished to record and quantify engine response characteristics. These quick stops were performed at 50 to 75 feet above ground level under the constraints of the SOFR and a test requirement for constant altitude decelerations. Figure 200, appendix G, shows a quick stop initiated at 76 KCAS. The maneuver was quite abrupt. Engine torque stabilized at the minimum value for 1 second; however, a comparison of rotor speed and power turbine speed (Np) indicated no evidence of "needle split" (autorotation was not entered). Peak rotor speed was 299 rpm (103.5 percent). Minimum rotor speed was 271 rpm (94 percent), and rotor speed was below 100 percent for only 2.3 seconds. The No. 2 engine fuel flow characteristics show a lag behind No. 1 engine fuel flow, followed by a high peak fuel flow which quickly decreased to a close approximation of the No. 1 engine fuel flow. This characteristic may be caused in part by the different group HMU's on the two engines (para 141).

144. Figure 201, appendix G, is a quick stop entered at 76 KTAS and terminated in a topping power vertical climb. Engine dynamics did not differ significantly from the previously described maneuver. Torque matching was excellent throughout the power range and is an enhancing characteristic of the YT700-GE-700 engine, as previously mentioned in paragraphs 81 and 90.

145. Figure 202, appendix G, depicts a quick stop entered at 125 KTAS. This maneuver was more gentle than the 76 KTAS quick stop due to the constant altitude constraint. Power was at the minimum for 9.5 seconds and collective control was on the lower stop for 8.5 seconds. The relationship between power turbine speed and rotor speed shows that the aircraft was in autorotation during this maneuver. Torque matching was excellent. Engine response characteristics did not limit this maneuver or any others performed during this evaluation.

<u>Airspeed Calibration</u>

146. The airspeed calibration of the YAH-64 was accomplished by pacing the aircraft with a calibrated AH-1G helicopter. Aircraft SN 74-22248 was calibrated with and without the instrumentation boom installed. Calibration data are presented in figures 203 and 204, appendix G.

CONCLUSIONS

GENERAL

147. After completion of flight tests on the YAH-64 and examination of all data available, the following general conclusions were reached:

a. The YAH-64 has the potential to be developed into an excellent attack helicopter (para 7).

b. Numerous envelope limits were imposed during this evaluation which would be unacceptable for an operational aircraft.

c. Collective and lateral cyclic actuator structural load instrumentation had to be monitored during maneuvering flight to assure that control system loads were not exceeded.

d. Although the hover characteristics of the YAH-64 were degraded by directional control characteristics in crosswinds of 10 KTAS or more, the hover stability of the aircraft was excellent under less critical wind conditions (para 87).

e. Within the temperature ranges encountered during this test (50 to 110°F), the ECS was very effective (para 119).

f. During night OGE hovering flight the inclusion of a radar altimeter as test instrumentation greatly aided the pilot in establishing and maintaining precise hover altitude (para 87).

g. Although a number of shortcomings have been noted, the overall cockpit configuration is a significant improvement in attack helicopter design (para 119).

h. The following enhancing characteristics were identified:

(1) The excellent SAS OFF handling qualities (para 101).

(2) The low level of physiologically significant vibrations at the crew stations (para 112).

(3) Torque matching between the two engines throughout the engine torque range was excellent (para 81).

(4) The ability to rapidly attain and precisely control hover altitude (para 87).

(5) Tail wheel configuration (para 88).

(6) The SAS monitor system (para 100).

(7) The steady-state autorotational descent performance characteristics (para 26).

(8) The excellent rotor speed control in autorotational descent (para 106).

(9) The capability of the aircraft to safely taxi rearward (para 70).

(10) The minimal effect of the failure of a single generator (para 102).

(11) The flashing caution panel effectively emphasizes systems problems (para 116).

(12) The ARTSS, controlled by a button on the pilot cyclic stick, allowed the pilot to individually select each of the three radio transmitters without removing his hands from the controls (para 85).

(13) The pilot left overhead circuit breaker panel provided quick identification and access to circuit breakers (para 116).

(14) The Marconi digital fiberoptic instrument presentation allowed rapid interpretation of parameters and the small multigauge panel located in the forward cockpit provided six individually selectable digital readout parameters (para 116).

(15) Incorporation of a rotor brake (para 116).

i. Seven deficiencies and 64 shortcomings were identified.

DEFICIENCIES

148. The following deficiencies (in order of their importance) were identified:

a. Loss of directional control in 25 to 30 KTAS left sideward flight (left hovering crosswind) (para 77).

b. Tail rotor horsepower exceeding the limit in right hovering crosswind (para 75).

c. The structurally inadequate tail rotor drive shaft-mounted cooling fan (para 132).

d. The unreliable APU starting system (para 130).

e. The unreliable air start system for the engines (para 131).

f. Lack of preset frequencies on the VHF and UHF radios (para 85).

g. The zero shifts of the Marconi fuel quantity gauges (para 135).

SHORTCOMINGS

149. The following shortcomings (in order of their importance) were identified:

a. The restricted field of view due to canopy structure and distortion (para 86).

b. Lack of foreign object protection boots where the directional controls pass through the floor (para 118).

c. The present configuration of the cockpits restricts the passing of essential maps or other mission-related information between the pilot and copilot (para 118).

d. The lower left circuit breaker panel cannot be adequately seen under either day or night conditions (para 127).

e. The poor braking characteristics at high ground speeds (para 69). N/C

f. The excessive brake pedal pressure required (para 69). N/C

g. The pitch-up from a hover to 60 KCAS requiring a 3-inch forward cyclic control input (para 71).

h. The excessive utility hydraulic accumulator manual recharging time (para 130).

i. The undesirable location of the weapons action switch (para 93).

j. The restricted forward field of view due to nose-high attitude (para 42).

k. The time delay in advacing the engine power lever due to engine oil pressure limits (para 83).

l. The intermittent Marconi gauge failure (para 136).

m. The position of the power levers in the OFF position blocks pilot and copilot view of certain fuel management switches (para 118).

n. Canopy drumming in the aft cockpit increases pilot fatigue (para 118).

o. The copilot canopy door handle cannot be reached by a 95th percentile pilot when secured in his seat (para 118).

p. The inability to read the lighted caution panel segments in bright sunlight (para 118).

q. Lack of quick access to the interior of the forward avionics bays (para 114).

r. The unreliable engine beep trim system (para 81).

s. The inability to identify a tripped circuit breaker on either circuit breaker panel at night (para 127).

t. The pitch-to-sideslip coupling (para 54).

u. The lack of a single-axis trim capability (para 99).

v. The wide dispersion of the basic flight instruments (para 98).

w. The intermittent binding of the tail wheel locking pin (para 69).

x. The pilot and copilot directional control pedal adjustment knobs are poorly located and require an excessive number of turns to adjust pedal position (para 118).

y. The longitudinal static stability characteristics degraded aircraft trimmability at airspeeds greater than 60 KCAS when small airspeed changes (2 to 4 knots) were attempted (para 41).

z. The directional control jump when retrimming (para 33).

aa. Weapon gas fumes in the cockpit during weapons firing (para 94).

bb. Cockpit entrance step mountings are too weak (para 118).

cc. Lack of pilot and copilot longitudinal seat adjustment (para 118).

dd. Pilot and copilot cockpit entrance and egress hindered by the cyclic stick (para 118).

ee. Seat vertical adjustments are difficult to operate (para 118).

ff. The lack of floor ramps (heel rests) under the copilot directional control pedals (para 118).

gg. Excessive dimming of the master caution light (para 122).

hh. Lack of light intensity control for the red and white floodlights (para 126).

ii. Lack of a magnetic compass in the forward cockpit (para 118).

jj. The longitudinal control shift in left sideward flight (para 76).

kk. The large longitudinal cyclic trim change due to power variation (para 43).

ll. Poor longitudinal cyclic control centering after aft control displacement (para 32).

mm. The lack of cyclic control harmony (para 67).

nn. The siphoning of fuel from the forward tank to the aft tank (para 134).

oo. An excessive amount of oil is thrown out by the engines (para 137).

pp. The unreliable low oil pressure switches for the main and accessory gearboxes (para 133).

qq. The common gyro for the pilot VSD and turn needle (para 118).

rr. Lack of a turn needle in the forward (copilot/gunner) cockpit (para 118).

ss. The reflection of the pilot map light by the blast shield (para 123).

tt. The copilot floor radio/intercom switch is awkward to use due to its mounting angle and location (para 118).

uu. Poor mounting of the pilot instrument panel ECS vents (para 118).

vv. The APU start switch can be inadvertently activated when climbing into or out of the pilot station (para 118).

ww. Lack of adequate secure storage space in the cockpits (para 118).

xx. The rotor brake can be inadvertently locked with the switch guard in place (para 118).

yy. After prolonged ground operation with the engines shut down and with electrical power applied to the aircraft by ground power units or the APU (*ie*, a large number of caution lights illuminated), the caution panel becomes extremely hot (para 118).

zz. The inability to check the main transmission oil sight gauges without removing the side panels (12 fasteners) or using an inspection mirror (para 114).

aaa. Lack of proper steps above the wing (para 114).

bbb. The forward engine door fasteners can be inadvertently unlatched by stepping on them while inspecting the rotor head (para 114).

ccc. The piano-hinge pins on the catwalk doors are not positively secured in the hinges (para 114).

ddd. The bolts securing the rotor blade strap packs to the rotor head are mounted with the nuts on top (para 114).

eee. Lack of sufficient lighting on the APU and engine oil sight gauges (para 114).

fff. The APU oil sight gauge is too sensitive to aircraft attitude (para 114).

ggg. The lead-acid battery venting outlet is located close to the external power receptacle (para 114).

hhh. Transmission compartment access doors cannot be latched open (para 114).

iii. The utility hydraulic reservoir sight gauge is obscured by the backup control system detent spring on the lateral control actuator (para 114).

jjj. The fluid level markings on the utility reservoir are decals and are subject to deterioration (para 114).

kkk. The difficulty in reading the primary hydraulic reservoir fluid level (para 114).

lll. Loss of continuous electrical power when transitioning from aircraft power to ground power (para 114).

SPECIFICATION COMPLIANCE

150. The YAH-64 was found to be not in compliance with the following paragraphs of the Army systems specification against which it was evaluated. Additional specification noncompliances beyond the scope of this evaluation may exist.

a. 3.2.1.1.1.a - The computed vertical climb rate was 184 ft/min, 266 ft/min less than specification (para 13).

b. 3.2.1.1.1.1b - The level flight cruise airspeed at MCP was 141 KTAS, 4 KTAS slower than specification (para 18).

c. 3.2.1.1.1.3b - The single-engine service ceiling was 4750 feet with the left engine operating and 4650 feet with the right engine operating; these values are 250 and 350 feet less than specification (para 15).

d. 3.7.1.4 - In that the rotor brake could be inadvertently placed in the lock position with the switch guard in place (para 117).

e. 10.3.2.1.1 - In that the aft breakout force (plus friction) was 1.8 pounds, 0.3 pound in excess of the specification limit (para 34).

f. 10.3.3.2.2 - In that objectionable control jump was experienced when retrimming the directional control (para 33).

g. 10.3.4.1 - A static longitudinal instability exists about a 113-KCAS trim airspeed (para 44).

h. 10.3.4.1.2 - Aircraft pitch attitude becomes more nose-up with increasing airspeed between 45 and 60 KCAS (para 44).

i. 10.3.6.3 - Objectionable pitch-to-sideslip coupling was noted (para 54).

j. 10.3.9.1.1 - It was not always possible to stabilize at left lateral airspeeds of 30 KTAS or more (para 84).

k. 10.3.9.1.2 - Insufficient longitudinal control margin exists to produce a 15-deg/sec pitch rate in 1.5 seconds in translational flight to 35 KTAS in any direction relative to the nose of the aircraft (para 78).

RECOMMENDATIONS

151. The deficiencies identified during this evaluation must be corrected for the YAH-64 to safely perform the attack helicopter mission (para 148).

152. The shortcomings identified during this evaluation should be corrected in the next development phase of the YAH-64 (para 149).

153. A radar altimeter should be installed in the production version of the YAH-64 (para 87).

154. The effect of compressibility during flights at high values of thrust coefficient should be evaluated during follow-on testing of the YAH-64.

APPENDIX A. REFERENCES

1. Test Plan, USAAEFA, Project No. 74-07, *Development Test I, Advanced Attack Helicopter Competitive Evaluation, YAH-63 Helicopter, YAH-64 Helicopter,* May 1976.

2. Systems Specification, United States Army Material Command, AMC-SS-AAH-10000A, "Systems Specification for Advanced Attack Helicopter," 1 July 1976.

3. Systems Specification, Hughes Helicopter, AMC-SS-AAH-H10000, "Systems Specification for Advanced Attack Helicopter," June 1973, as amended.

4. Letter, AVSCOM, DRSAV-EQ, 31 May 1976, subject: Safety-of-Flight Release for the YAH-64 Government Competitive Test, with Revision 1, 30 June 1976; Revision 2, 20 July 1976; and Revision 3, 16 August 1976.

5. Message, AVSCOM, DRSAV-EQ, 161815Z, August 1976, subject: AAH Government Competitive Test.

6. Engineering Design Handbook, Army Material Command, AMCP 706-204, *Helicopter Performance Testing,* August 1974.

7. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Stability and Control,* 10 June 1976.

8. Flight Test Manual, Naval Air Test Center, FTM No. 102, *Performance,* 28 June 1976.

9. Training Manual, Hughes Helicopter Company, *Hughes YAH-64 Advanced Attack Helicopter,* 1 April 1976 (CS).

10. Air Force Pamphlet, Department of the Air Force, AFP 161-16, *Physiology of Flight,* 1 April 1968.

11. Letter, AVSCOM, DRSAV-EQP, 26 August 1976, subject: GCT Exhaust System Performance Losses Using the T700 Engine Deck No. 73004.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The YAH-64 advanced attack helicopter is a tandem, two-place twin turbine-engine, single-main-rotor aircraft manufactured by Hughes Helicopter Company, a division of Summa Corporation. The aircraft is designed to deliver various combinations of ordnance stored both internally and externally on the four wing store positions during day and night combat conditions. Photos 1 through 4 are views of the prototye YAH-64. A three-view drawing of the YAH-64 is shown in figure 1. Basic design information is listed below. A complete description of the aircraft is contained in reference 9, appendix A. Major features of the helicopter are described below.

Dimensions and General Data

Main Rotor:	
Diameter	48 ft
Chord constant	21 in.
Main rotor blade area	150.88 ft^2
Main rotor disc area	1809.55 ft^2
Main rotor solidity	0.092
Airfoil	HH-02
Twist	9 deg washout
Number of blades	4
Rotor speed at 100% Np	289 rpm
Normal tip speed (ΩR)	726.36 ft/sec
Tail rotor:	
Diameter	8.33 ft
Chord constant	10 in.
Tail rotor blade area	10 ft^2
Tail rotor disc area	54.54 ft^2
Tail rotor solidity	0.2475
Airfoil	NACA 632-414
Twist	8 deg washout
Number of blades	4
Rotor speed at 100% Np	289.3 rpm
Distance from main rotor	28.49 ft
Normal tip speed (ΩR)	727.09 ft/sec

Wing:

Span	16.33 ft
Chord (mean aerodynamic)	45.9 in.
Total area	61.56 ft^2
Flap area	8.71 ft^2
Airfoil at root	NACA 4418
Airfoil at tip	NACA 4415.5
Thickness (avg)	16.9% chord
Incidence	6 deg
Aspect ratio	4.26

Weights:

Empty weight	10,495 lb (SOFR)
Design gross weight	14,242 lb (calculated)
Maximum gross weight	17,900 lb (SOFR)

AIRFRAME

2. The airframe, as shown in figure 2, includes the nose section, avionics bays, cockpit, center fuselage section, tail boom, vertical stabilizer, and horizontal stabilizer. The fuselage is of a semimonocoque construction of primarily aluminum alloys. It consists of 10 major bulkheads and frames and 8 major longerons and stringers.

3. The cockpit canopy is constructed of seven sections of shatter-resistant plexiglass, some of which are slightly curved to reduce cockpit drumming. The windshields are air-heated for defogging and anti-icing. The canopy is equipped with an explosive jettison device, activated from either cockpit or from an externally accessible trigger in the nose of the aircraft. Electrically operated windshield wipers are provided on two windshields, one forward of and another directly over the copilot/gunner crew station. Dual controls and some duplication of flight instruments are provided. Cockpit drawings are presented in figures 3 and 4. The pilot in the rear is separated from the front cockpit by a transparent blast shield and a nontransparent lower crew barrier. The pilot sits approximately 3 feet forward of the main rotor and 19 inches higher than the copilot/gunner (variable with seat height adjustment).

4. The center fuselage section connects the cockpit section with the tail boom structure and contains the two main bladder-type fuel cells, one each located fore and aft of the ammunition bay. The forward fuel cell capacity is 151 gallons (981.5 pounds), and the aft cell holds 202 gallons (1313 pounds). The center fuselage section also mounts the wing, which is of three-spar tapered box beam construction with multiple ribs. The main rotor transmission mounts atop the center fuselage section, and one each of the two engines mounts high and to either side of this section. The engine nacelle doors open to form work platforms which are designed for two mechanics each (photo 5).

Photo 1. Front View.

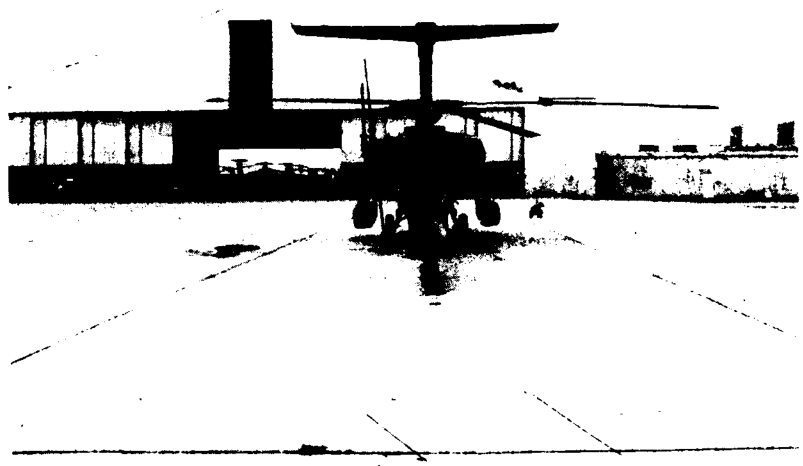


Photo 2. Rear View.

Photo 3. Left Side View.

Photo 4. Left Front Quarter View.

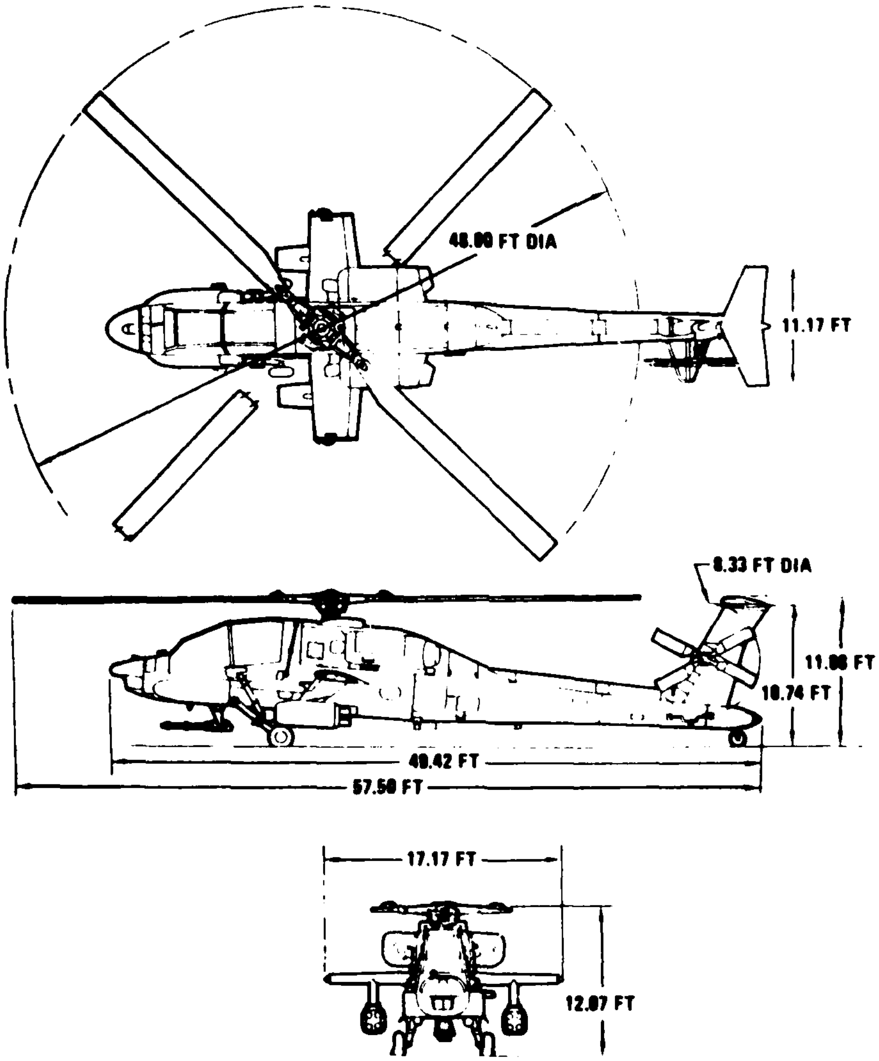


Figure 1. Dimensions.

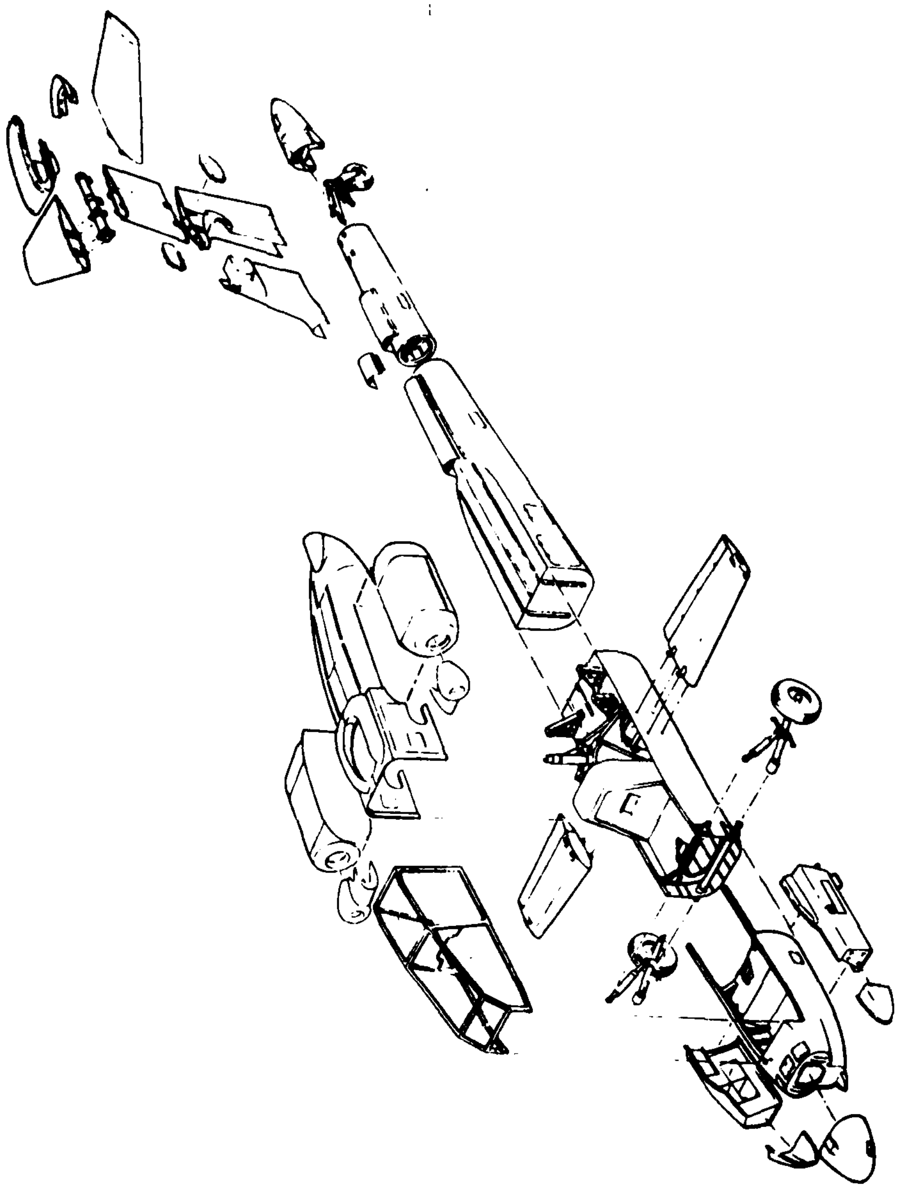


Figure 2. Airframe.

C.P.G. STATION

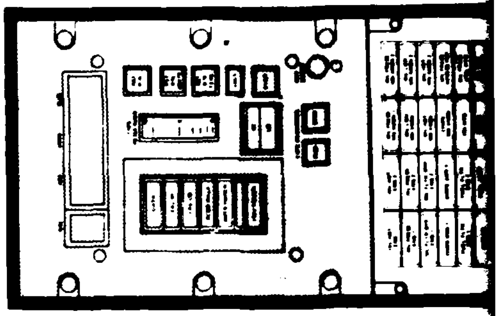
1. AIRSPEED *
2. SENSITIVE ALTIMETER
3. STANDBY ATTITUDE
4. IVSI
5. FUEL QUANTITY; FWD, AFT
6. Ng
7. ENGINE AND TOTAL TORQUE
8. Np AND Nr
9. ENGINE INSTRUMENT PRE-FLIGHT TEST SWITCH
10. FIRE DETECTION SYSTEM TEST
11. FUEL FLOW AND TOTALIZER; LEFT AND RIGHT ENGINE *
12. CONDITIONED AIR OUTLETS
13. WARNING PANEL
14. EMERGENCY CANOPY JETTISON
15. FIRE EXTINGUISHER PANEL
16. ARMAMENT PANEL
17. WEAPON CONTROL PANEL
18. EMERGENCY STORES JETTISON
19. T 4.5 INDICATORS; LEFT AND RIGHT *
20. SHIP'S AIRSPEED *
21. ANGLE OF SIDESLIP *

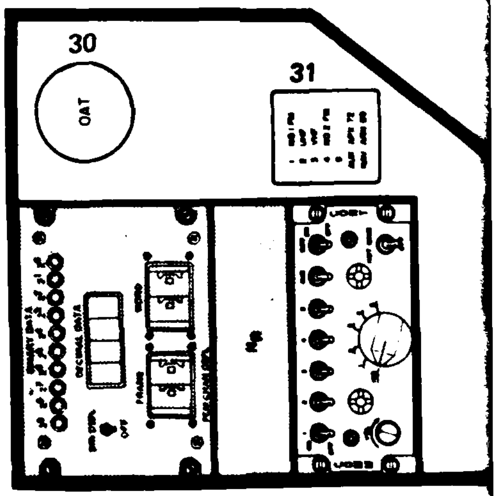
C.P.G. CONSOLES

22. P.C.M. DISPLAY *
23. Nr SENSITIVE *
24. ICS
25. POWER QUADRANT
26. EMERGENCY FUEL CONTROL
27. INTERIOR LIGHTING CONTROLS
28. ANTI-ICE
29. STOWAGE
30. OAT *
31. RADIO PLACARD
32. COLLECTIVE STICK GRIP
33. INSTRUMENTATION CONTROL PANEL *
34. CAUTION PANEL
35. CIRCUIT BREAKERS
36. COLLECTIVE FRICTION CONTROL
37. HAND HELD FIRE EXTINGUISHER

NOTE: * DENOTES FLIGHT TEST EQUIPMENT

33





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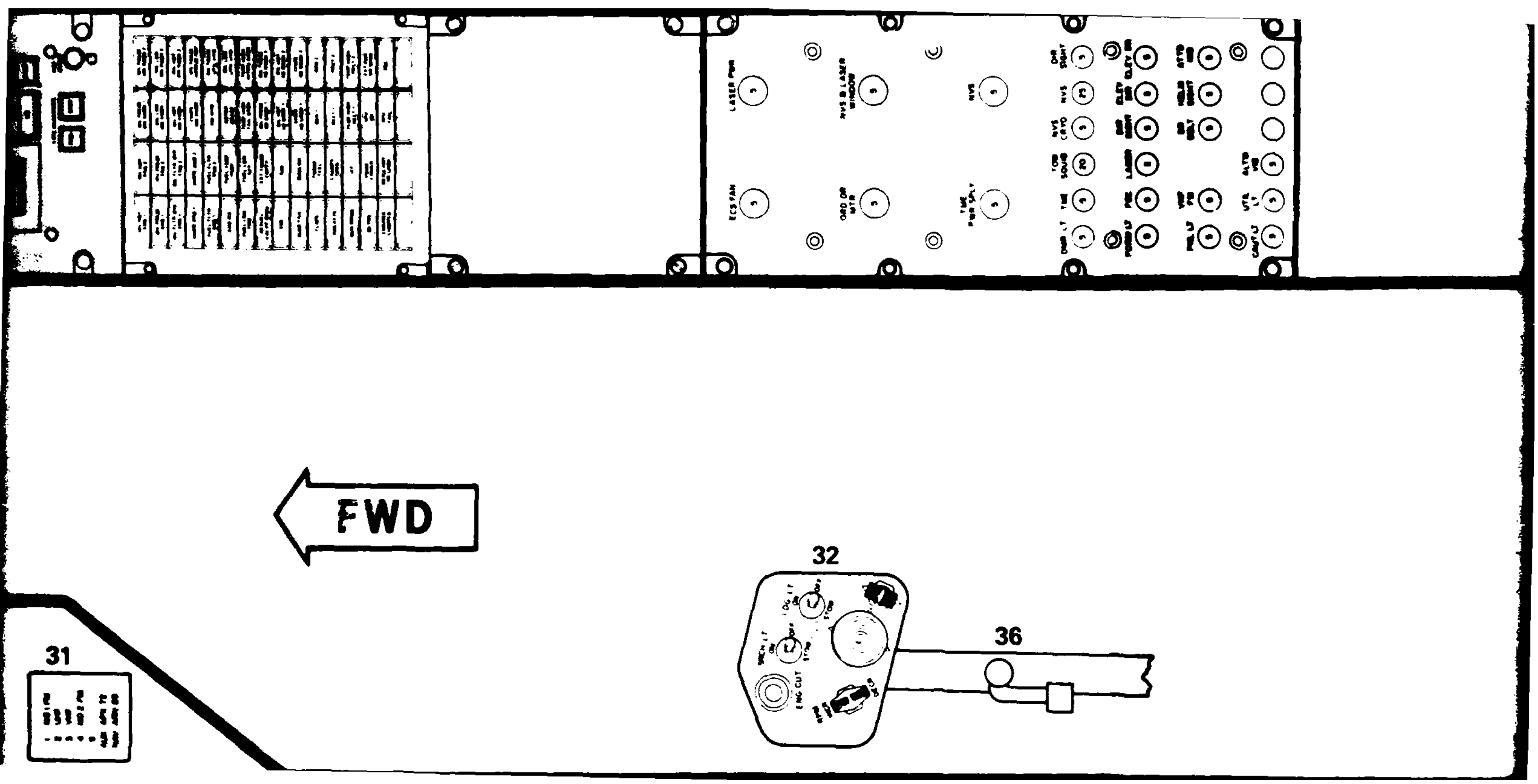
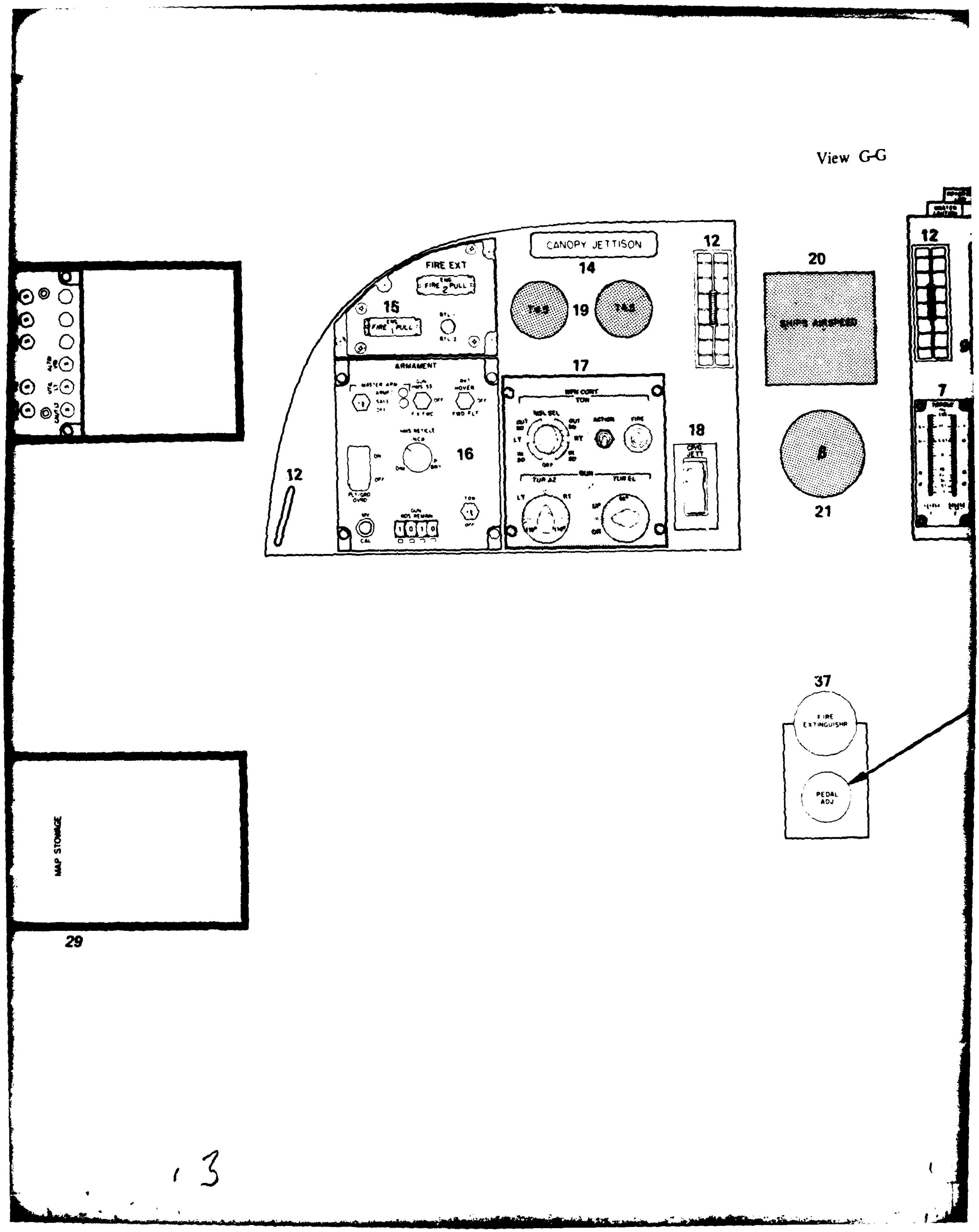
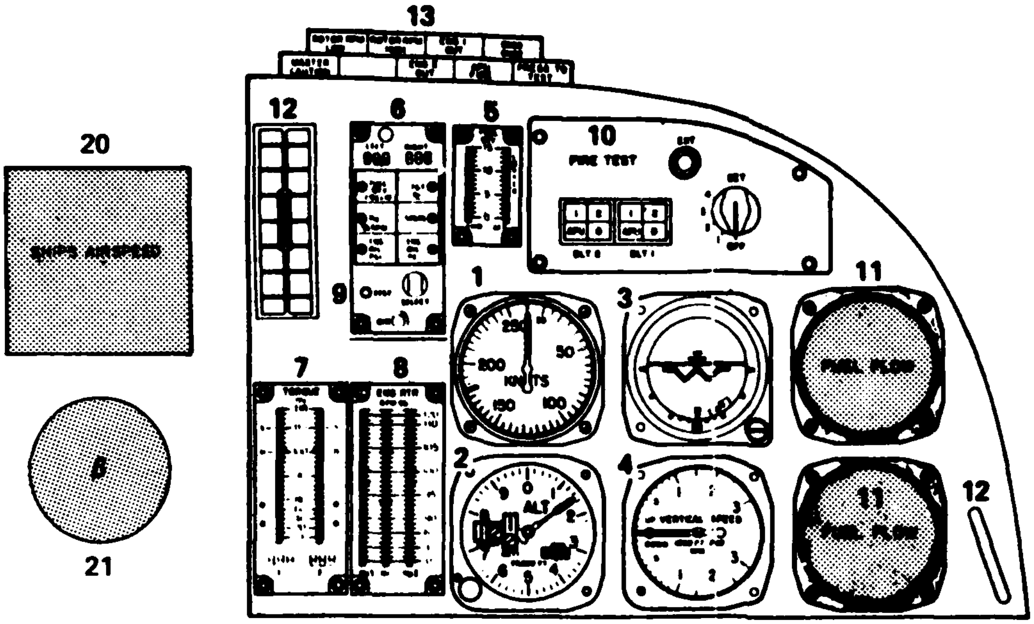
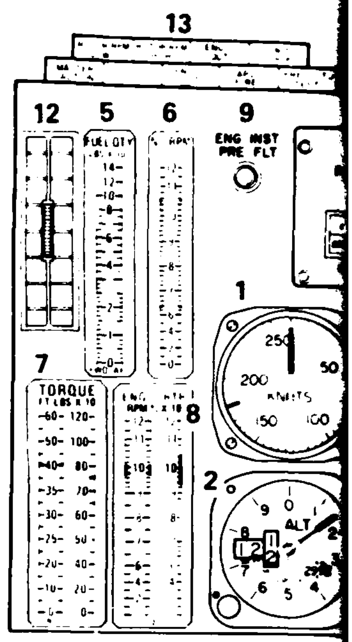

FWD
32
36
31

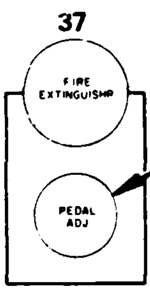

View G-G
CANOPY JETTISON
FIRE EXT
ARMAMENT
MAP STOWAGE
FIRE EXTINGUISHR
PEDAL ADJ
12
14
15
16
17
18
19
20
21
29
37
7
9

View G-G

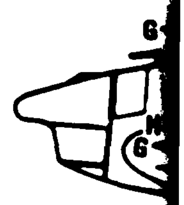




Gage P



Anti-torque
foot pedal
adjustment
knob

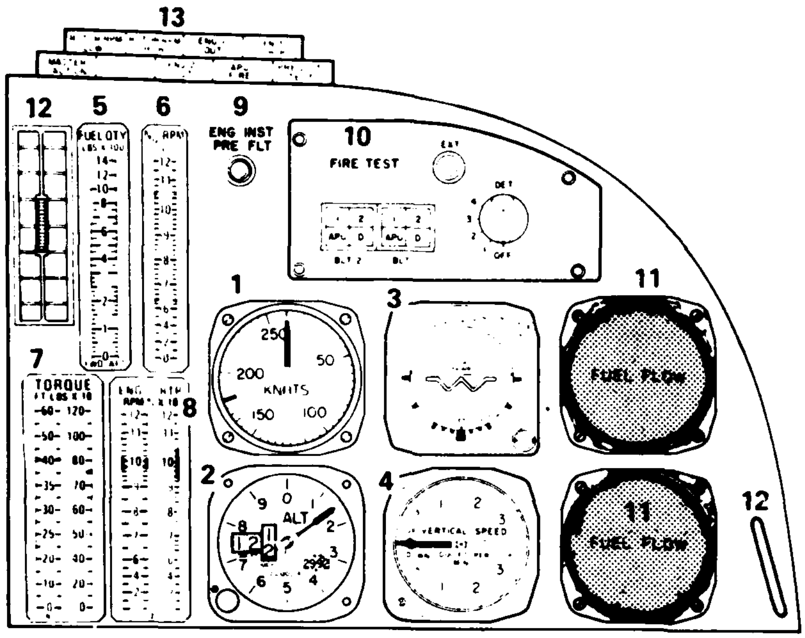


Note: Tow stabilized sight can be installed if β and ships ASI are removed.

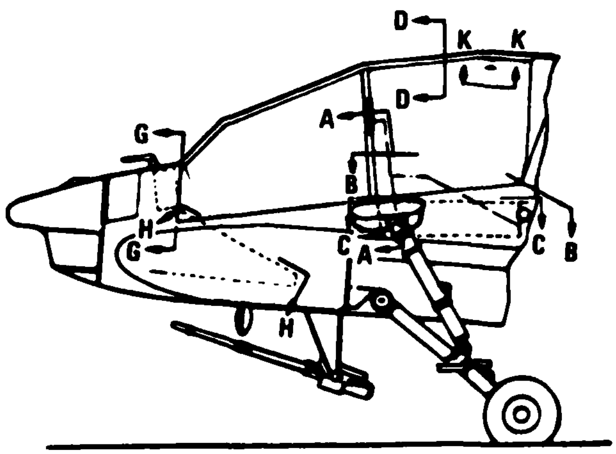
Shaded instruments represent flight test equipment.

NOTE: AL

4



Gage Presentation for Aircraft
SN 74-22248



NOTE: ALL VIEWS SHOWN IN THE PLANE OF PROJECTION

Figure 3. Copilot Cockpit.

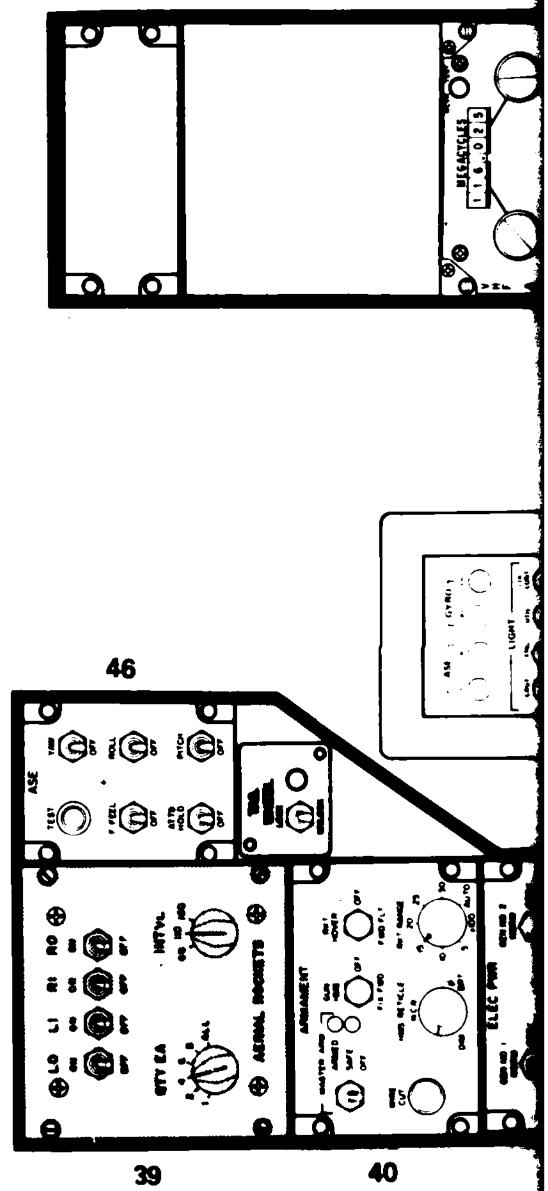
PILOT'S STATION

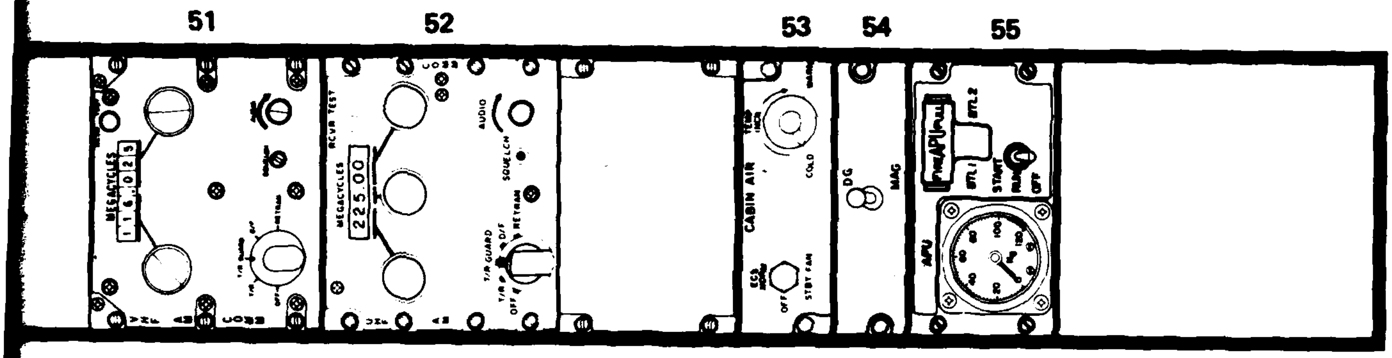
1. VERTICAL SITUATION DISPLAY (VSD)
2. AIRSPEED
3. STANDBY ATTITUDE INDICATOR
4. ALTIMETER
5. IVSI
6. ACCELEROMETER
7. ENGINE AND TOTAL TORQUE
8. Np AND Nr
9. FUEL QUANTITY; FWD, AFT, TOTAL
10. ENGINE OIL PRESSURE
11. ENGINE OIL TEMPERATURE
12. TURBINE GAS TEMPERATURE (TGT)
13. Ng
14. A.C. LOADMETER
15. D.C. AMMETER
16. HYDRAULIC PRESSURE INDICATOR
17. EMERGENCY HYDRAULICS SWITCH
18. STORES JETTISON CONTROL
19. EMERGENCY CANOPY JETTISON
20. CONDITIONED AIR OUTLETS
21. ASN/43 DISPLAY*
22. INTERCOM. (ICS)
23. INSTRUMENTATION CONTROL PANEL*
24. VSD CONTROL
25. RADIO PLACARD
26. SENSITIVE Nr* (DIGITAL)
27. FOUR AXIS STICK POSITION INDICATORS*
28. SIDESLIP ANGLE*
29. CLOCK
30. CAUTION PANEL
31. WARNING PANEL
32. *FIRE EXTINGUISHER* BOTTLES 1 AND 2 SELECT SWITCH
33. ENGINE FIRE WARNING PULL HANDLES
34. ENGINE INSTRUMENTS PRE-FLIGHT TEST SWITCH
35. OUTSIDE AIR TEMPERATURE (OAT)
36. MAGNETIC COMPASS
37. GENERATOR 1 CIRCUIT BREAKERS
38. GENERATOR 2 CIRCUIT BREAKERS

38A. RATE OF TURN AND INCLINOMETER PILOT'S CONSOLES
38B. V_{NE} PLACARD*
38C. TORQUE LIMIT PLACARD*
38D. LATERAL ACTUATOR LOAD INDICATOR*
38E. AUTOMATED RADIO TRANSMIT SELECT SYSTEM DISPLAY*

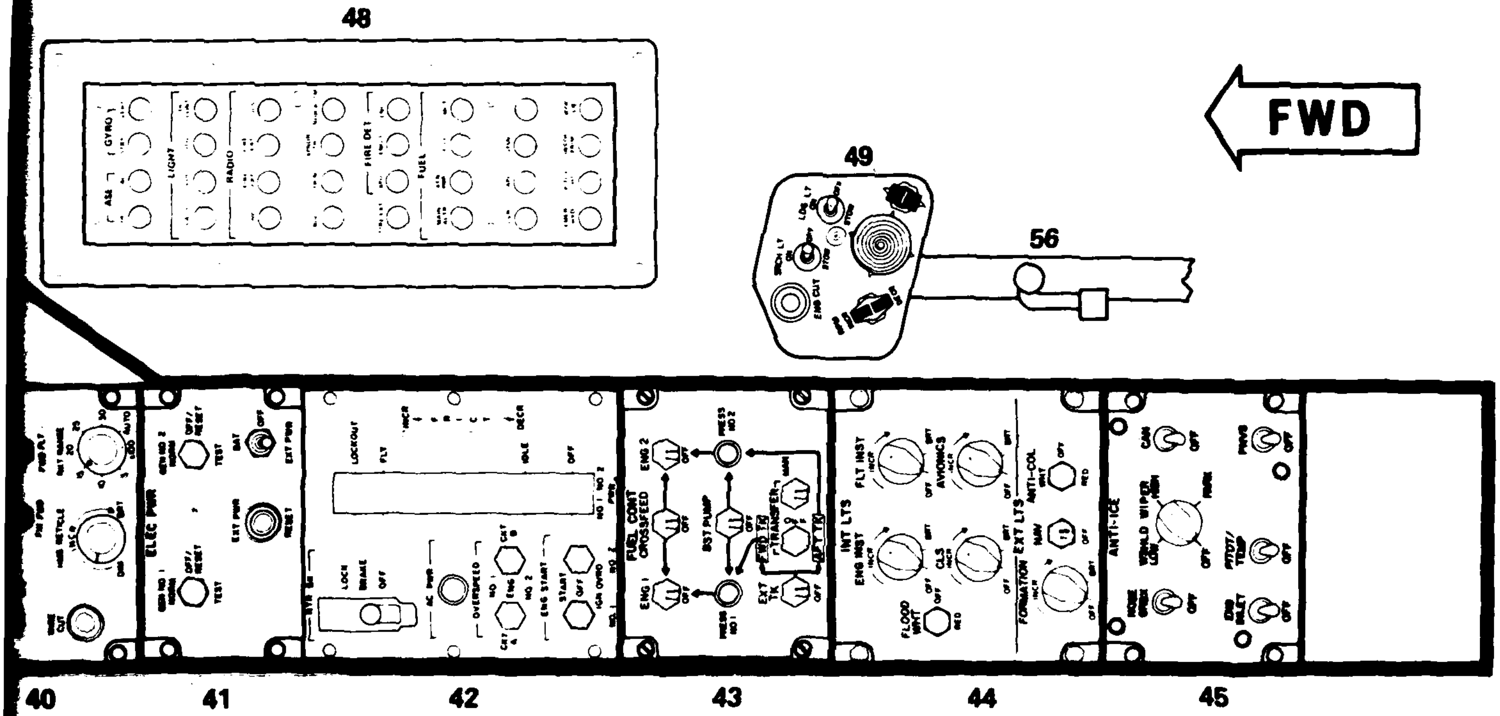
39. ROCKET CONTROL PANEL*
40. ARMAMENT MASTER CONTROL
41. ELECTRICAL CONTROL PANEL
42. POWER QUADRANT
43. FUEL CONTROL PANEL
44. INTERIOR AND EXTERIOR LIGHTING CONTROLS
45. ANTI-ICE AND WINDSHIELD WIPERS
46. AUTOMATIC STABILIZATION EQUIPMENT (ASE) CONTROL
47. TAIL WHEEL LOCK
48. EMERGENCY BUS CIRCUIT BREAKERS
49. COLLECTIVE STICK GRIP
50. DELETED
51. ARC 114 VHF-AM RADIO
52. ARC 115 UHF-AM RADIO
53. CABIN AIR CONTROL
54. ASN/43 CONTROL*
55. AUXILIARY POWER UNIT (APU) CONTROL AND FIRE HANDLE
56. COLLECTIVE FRICTION CONTROL

NOTE: *DENOTES FLIGHT TEST EQUIPMENT.

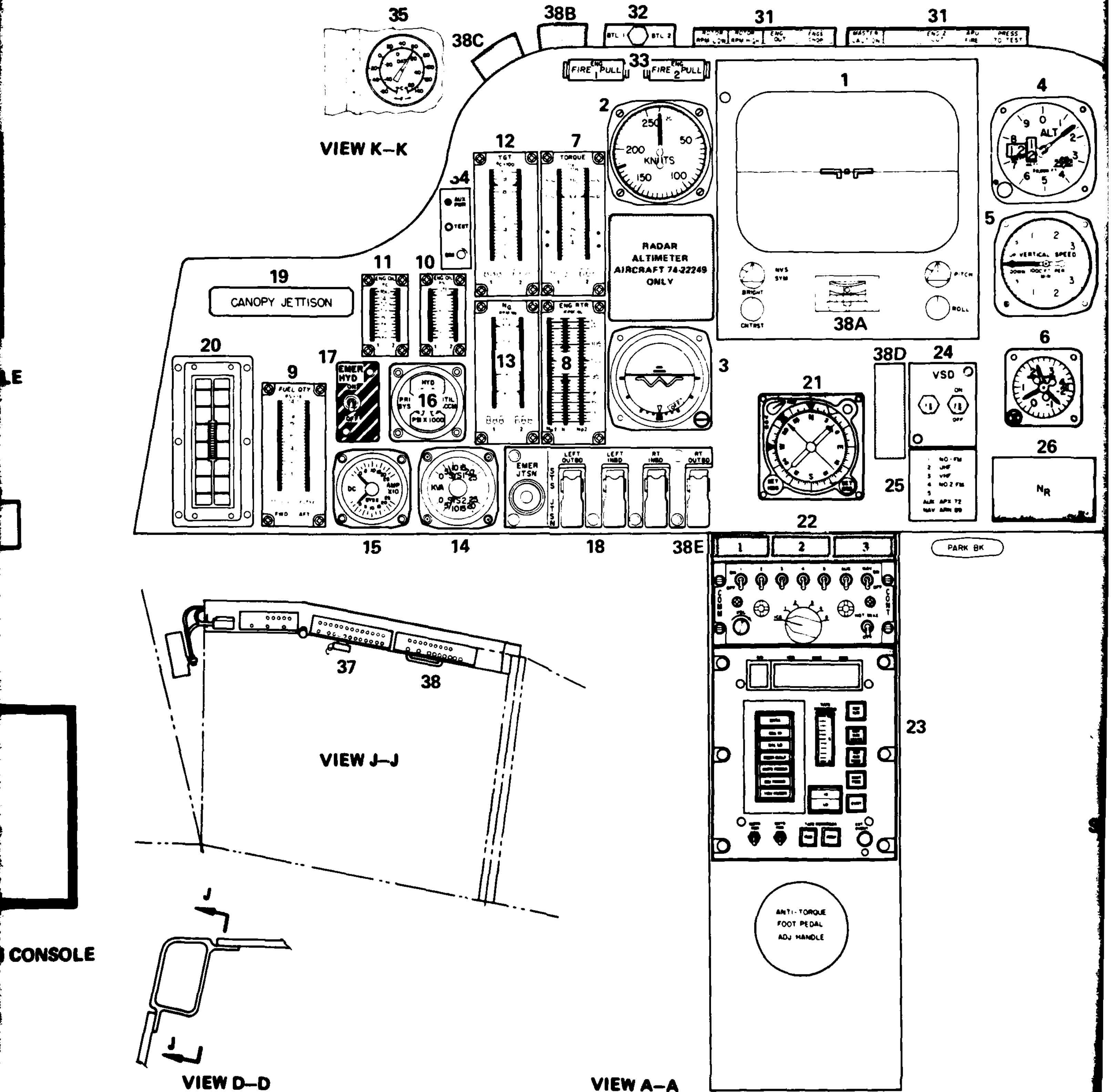




VIEW C–C RH CONSOLE



VIEW B–B LH CONSOLE



VIEW K–K
VIEW A–A
VIEW J–J
VIEW D–D
CONSOLE
LE
CANOPY JETTISON
RADAR ALTIMETER AIRCRAFT 74-22249 ONLY
FIRE ENG 1 PULL
FIRE ENG 2 PULL
BTL 1
BTL 2
KNOTS
ALT
VERTICAL SPEED
TGT
TORQUE
ENG OIL
ENG RTR
FUEL QTY
FWD
AFT
EMER HYD
HYD
PSI X 1000
AMP X10
DC
KVA
EMER JTSN
LEFT OUTBD
LEFT INBD
RT INBD
RT OUTBD
VSD
ON
OFF
AUX PWR
TEST
BRIGHT
CNTRST
PITCH
ROLL
NVS SYM
N_R
PARK BK
ANTI-TORQUE FOOT PEDAL ADJ HANDLE
COMM
CONT
1 NO 1 FM
2 UHF
3 VHF
4 NO 2 FM
5
AUX APX 72
NAV ARN 89
1
2
3
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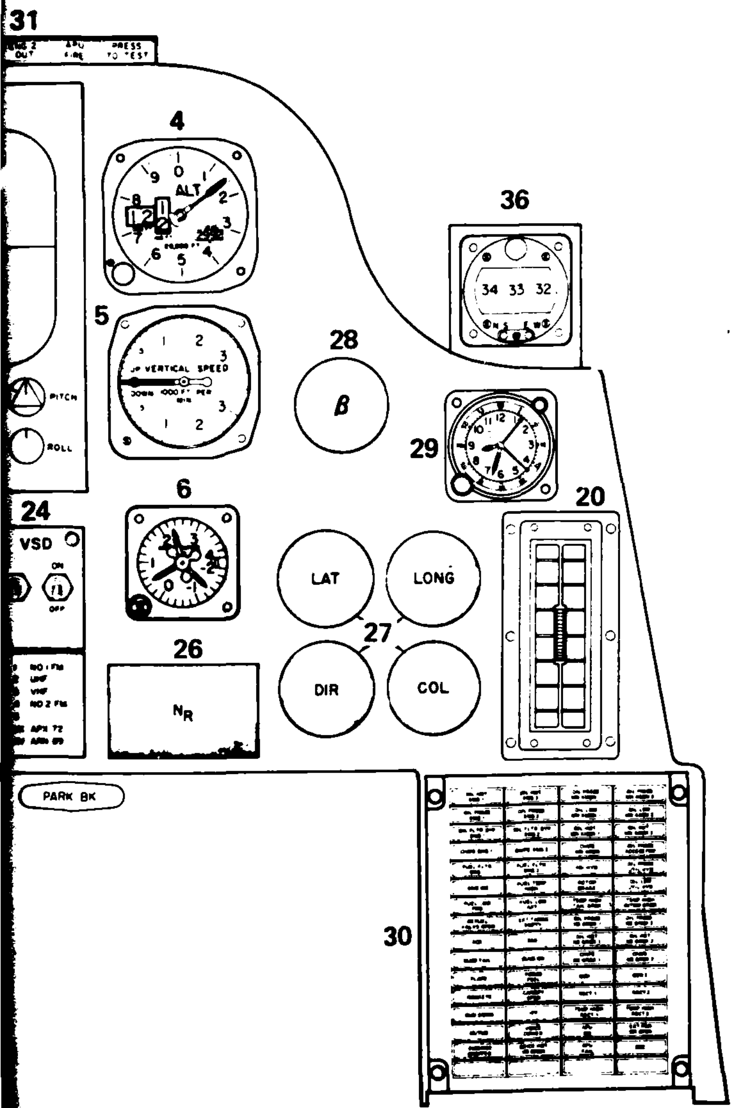
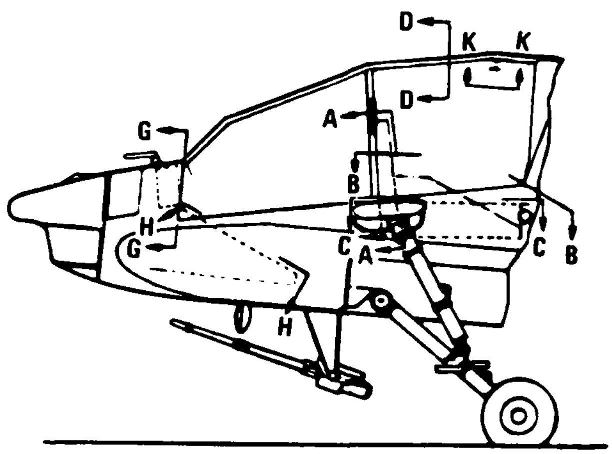


SHADED INSTRUMENTS REPRESENT
FLIGHT TEST EQUIPMENT



NOTE: ALL VIEWS SHOWN IN THE PLANE OF PROJECTION

FOR OFFICIAL USE ONLY

Figure 4. Pilot's Crew Station.

4

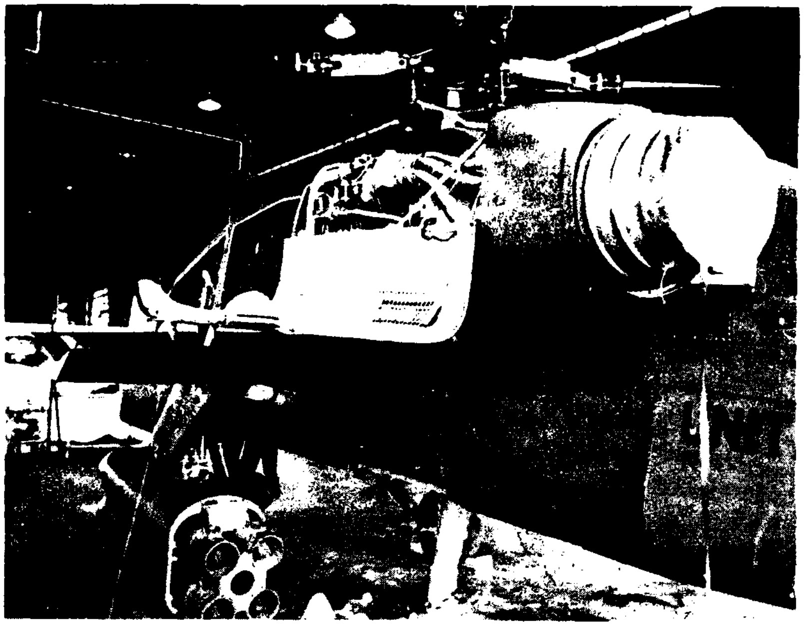


Photo 5. Left Engine Work Platform.

5. The tail boom houses the tail rotor drive shaft, tail rotor controls, and numerous electrical wiring bundles. It also provides a mounting point for the grease-lubricated intermediate gearbox and the swiveling tail wheel. The tail boom is a semimonocoque structure consisting of a basic circular section and twelve longerons. The skin of the tail boom varies from 0.025 to 0.032 inch thick. The vertical stabilizer is mounted atop the rear section of the tail boom. It is cambered to unload the tail rotor in cruising flight. The vertical stabilizer is of a tapered box beam construction, with two main spars and two intermediate spars. The tail rotor and fixed horizontal stabilizer are mounted on the side of and atop the vertical stabilizer. The horizontal stabilizer is positioned for pitch attitude control in forward flight; it is of a bonded sandwich construction employing stressed skin.

MAIN ROTOR

6. The main rotor is a fully-articulated, four-bladed system mounted on a static mast which carries the vertical loads and bending moments directly into the mast support structure. A drive shaft is mounted coaxially within the static mast to transmit torque to the main rotor. The main rotor blade airfoil section is a modified cambered type with a full span reflexed trailing edge tab. The main rotor diameter is 48 feet, and the blade chord is 21 inches. A typical main rotor blade section is shown in figure 5. The blade structure is a four-cell box of laminated stainless steel and fiberglass construction. The stainless steel sections are reinforced by filament-wound inner fiberglass tubes. Other notable features of the main rotor

assembly include four V-shaped laminated stainless steel tension-torsion blade retention strap packs; eight elastomeric lead-lag dampers (valued at approximately 130,000 lb/in., that permit blade lead-lag motions); four adjustable pitch change links connected to the four pitch change housings; and the swash plate assembly and rotating scissors. The main rotor blades are attached to the rotor head (photo 6) by quick-release expandable bolts, two to each blade. A droop stop mechanism positioned around the static mast at the bottom of the hub assembly restricts downward flapping of the blades when the rotor is stopped.

Photo 6. Main Rotor Blade Attachment to Rotor Head.

TAIL ROTOR

7. The tail rotor system consists of the tail rotor head assembly and four rotor blades, mounted at an included angle of 55 degrees between spanwise axes (fig. 6 and photo 7). The head assembly consists primarily of two titanium hubs and two strap assemblies mounted in a titanium fork assembly. The blades are attached to the outboard ends of the stainless steel tension-torsion strap assemblies which pass through the hubs. The hub, strap assembly, and blades are attached to the fork assembly by a bolt which passes through elastomeric teeter bearings. The two

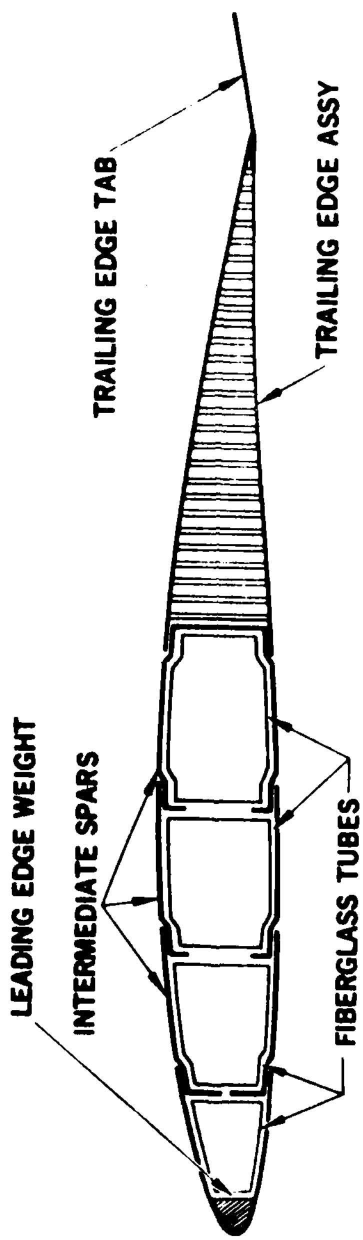


Figure 5. Main Rotor Blade Construction.

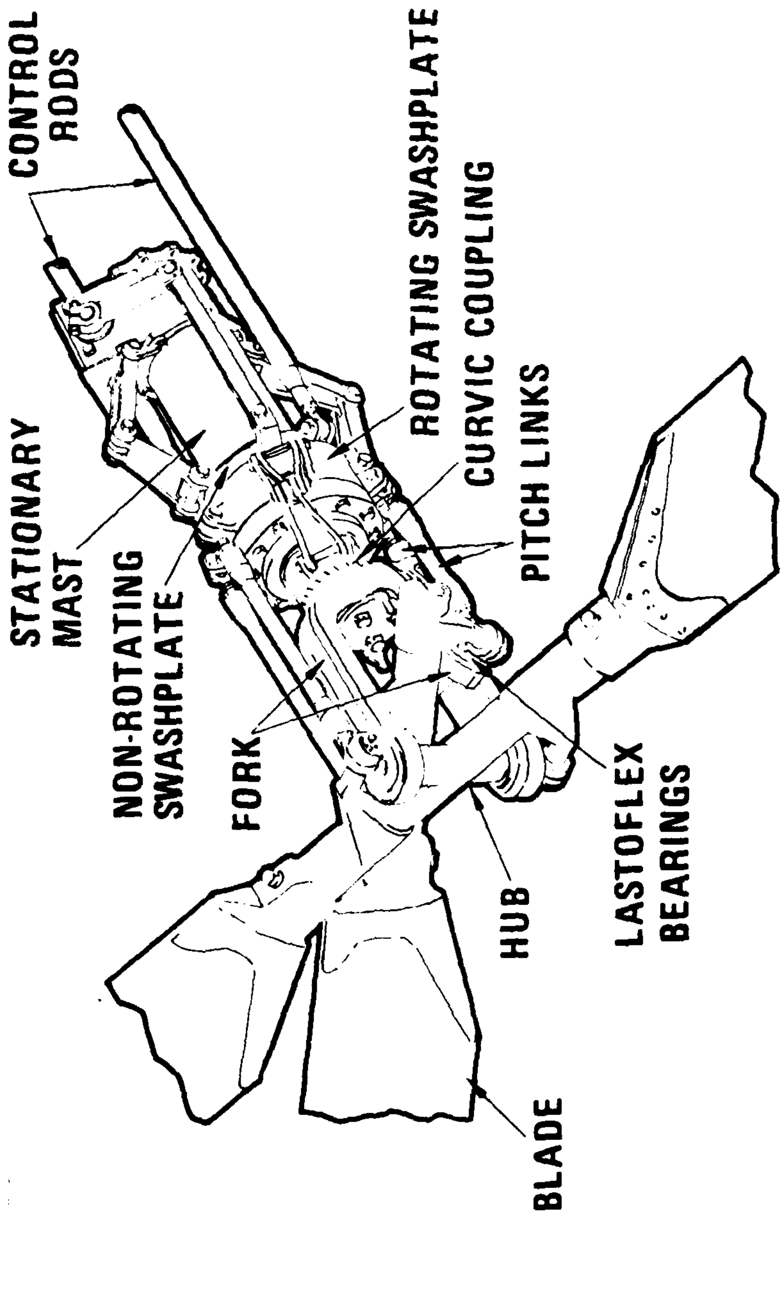


Figure 6. Tail Rotor System.

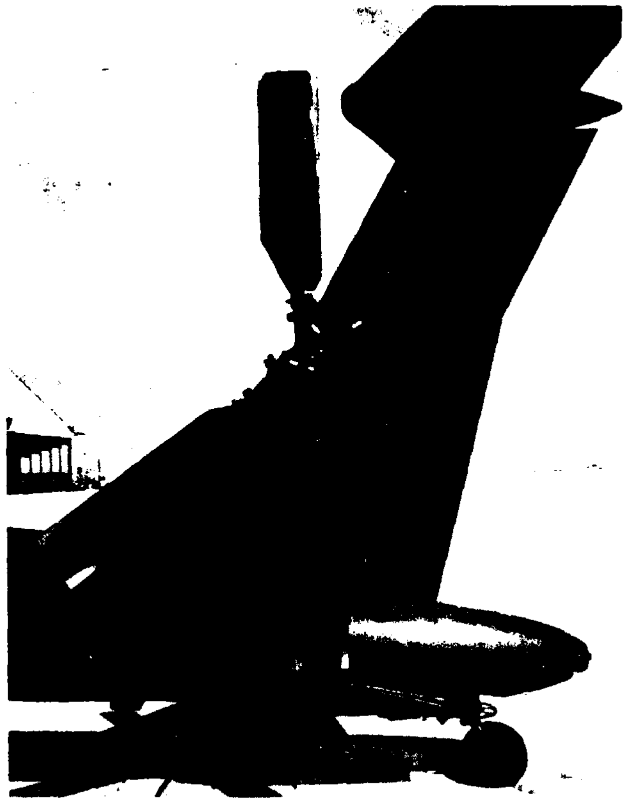


Photo 7. Tail Rotor System.

ROTOR BRAKE

8. An electrohydraulic rotor brake, located aft of the main transmission, is provided to stop rotation of the rotor systems. Additionally, the rotor brake may be used to hold the rotor system from turning during engine start and while the helicopter is parked. The brake is controlled by a guarded RTR BK switch on the pilot power quadrant (fig. 4). Pressure for the rotor brake is supplied by the utility hydraulic system. The rotor brake remains applied with the control switch in the LOCK position regardless of whether or not other aircraft systems are operating. However, electrical power is required to apply or unlock the rotor brake. Operating limitations require switching the rotor brake off prior to advancing the engine PCL's beyond the idle range, thus preventing damage to rotor brake components. Additionally, the rotor brake must not be engaged until both main engines have been shut down (N_R is less than or equal to 50 percent). To prevent incorrect operating procedures, electrical and mechanical interlocks have been

provided. The guard on the rotor brake control switch is designed to prevent inadvertently moving the switch to LOCK without first stopping the rotor system with the switch in the BRAKE position. A mechanical interlock physically prevents advancing either engine power level beyond the IDLE position unless the rotor brake control switch is OFF. Once the rotor brake has been switched off, an electrical interlock prevents its reactivation unless both PCL's are in the OFF position, regardless of the position of the rotor brake control switch. The RTR BRAKE caution light illuminates whenever hydraulic pressure is applied to the rotor brake caliper.

DRIVE SYSTEM

9. The drive system, shown in figure 7, consists of two engine nose gearboxes, a main transmission, an intermediate gearbox, a tail rotor gearbox, two engine nose gearbox to main transmission drive shafts, the main rotor drive shaft, and the tail rotor drive shafting.

Engine Nose Gearboxes

10. Engine nose gearboxes, mounted on the front of each engine, contain one set of spiral bevel gears that reduce nominal engine speed from 20,000 to 9841 rpm for input to the main transmission. A self-contained lubrication system which consists of a pressure pump, filter with bypass, and integral sump and oil passageways, is provided. Each gearbox has an oil level sight gauge, a chip detector, and temperature and pressure measuring systems. These systems activate caution lights in both crew stations when chips, high oil temperature, or low oil pressure are detected. An axial flow fan, mounted on the gearbox output shaft, draws air through the gearbox cowling past integral cooling fins and exhaust into the transmission compartment.

Main Transmission

11. The main transmission, mounted below the main rotor static mast base, couples the two engine nose gearbox inputs and provides outputs for the main rotor, tail rotor, accessories, and rotor brake disc. An overrunning clutch at each main transmission input permits either or both engines to be disengaged from the transmission during single-engine operation or autorotation. The main transmission reduces the 9841 rpm input to 289 rpm for the main rotor, 4842 rpm for the tail rotor drive, and 4757 rpm for the rotor brake disc.

12. The APU drives all accessories for subsystem checkout when the rotor is stopped. This is accomplished by the use of overrunning clutches in the accessory gear train at the tail rotor gear shaft and the rotor brake disc shaft. The accessories include two AC generators (alternators), two hydraulic pumps, three lubrication pumps, and a centrifugal compressor. The main transmission drives the accessories when they are not being driven by the APU. Main rotor speed is sensed by a magnetic pickup.

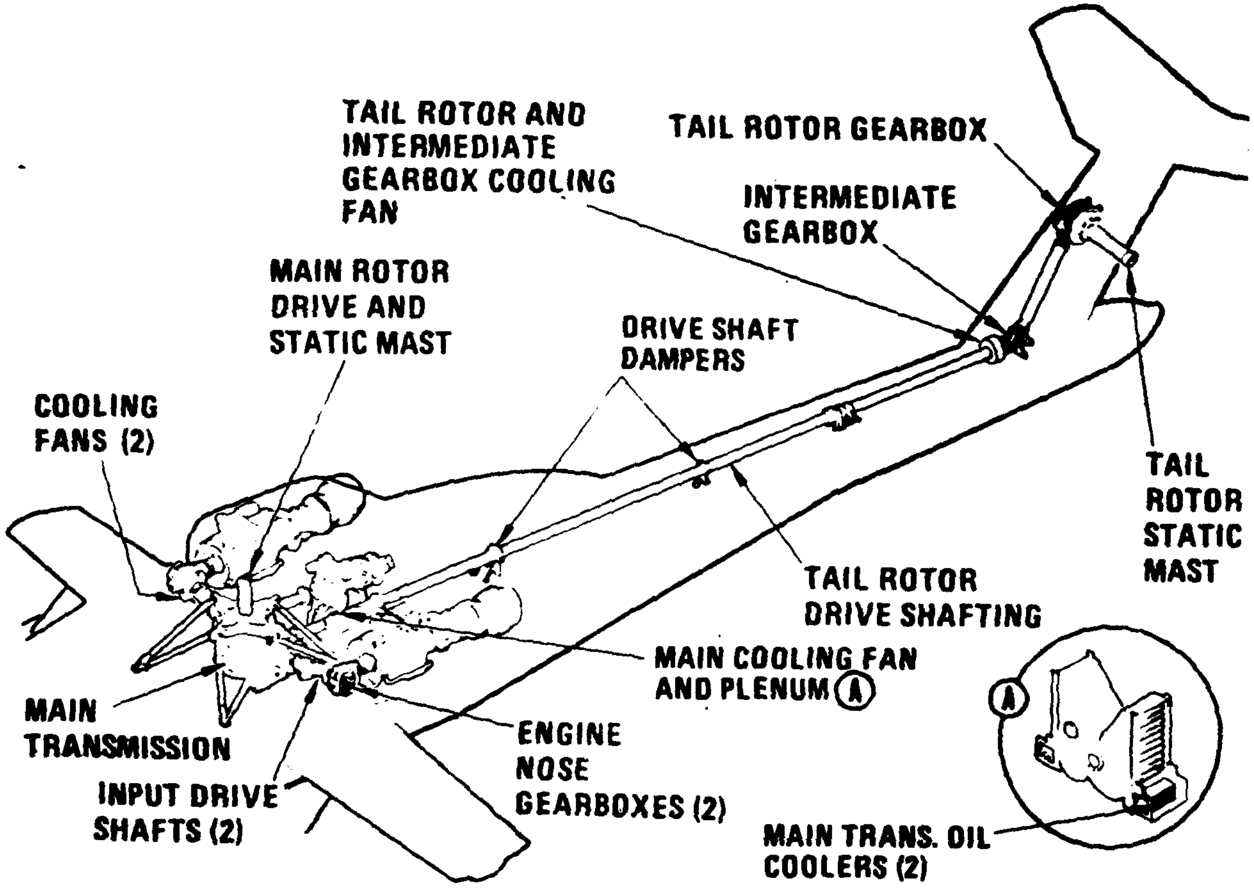


Figure 7. Drive Subsystem.

13. The main transmission features a dual lubrication system with each system consisting of a sump, pump, heat exchanger, filter with bypass, and associated valves and caution indicators. The two sumps have a capacity of 1.7 gallons each and are integral with the transmission housing. Oil level sight gauges are provided for each sump. The pumps are driven independently. An additional oil pump, driven by the accessory gear train, supplies oil to the accessory gear train and bearings when driven by the APU. The pump also provides lubrication for the centrifugal compressor. Heat exchangers for the main transmission lubrication system are located in the engine cooling fan plenum chamber. The heat exchangers have a pressure bypass valve to protect against cold start surge pressures. Air flow for these heat exchanges is provided by the engine cooling fan which is mounted on and driven by the tail rotor drive shaft. Exit air is ducted overboard. The transmission is also cooled externally with air drawn around the transmission housing by the engine cooling fan. Each lubrication system is monitored by caution lights for chips, main transmission and accessory gearbox low oil pressure, high oil temperature, and low oil level.

Intermediate Gearbox

14. The intermediate gearbox, located at the base of the vertical stabilizer, contains one set of spiral bevel gears which reduce the input speed from 4842 to 3656 rpm. The gearbox is grease lubricated and cooled by forced air. A fan mounted on the gearbox input shaft draws air from an inlet on the vertical stabilizer to provide cooling for both the tail rotor and intermediate gearboxes. Five thermistors are provided to monitor temperatures. A caution light illuminates in both crew stations when high temperatures are detected by any thermistor.

Tail Rotor Gearbox

15. The tail rotor gearbox, mounted on the vertical stabilizer, contains one set of spiral bevel gears. Input speed is reduced from 3656 to 1411 rpm. The tail rotor output shaft is contained in the static mast mounted on the gearbox. All tail rotor loads are transmitted by the static mast. The output shaft transmits only torque to the tail rotor. The gearbox is grease lubricated and cooled by forced air. Five thermistors are installed to sense high temperatures. High-temperature caution lights are provided in both crew stations.

Drive Shafts

16. The drive shafting consists of two engine nose gearbox to main transmission shafts, a main rotor shaft, and a four-section tail rotor drive shaft.

FUEL SYSTEM

17. The fuel system consists of two fuel cells within the fuselage, provisions for extended-range tanks, associated controls, plumbing, and pumps. The system

incorporates crashworthy features. Figure 8 presents a schematic diagram of the fuel system.

Main Fuel Cells and Extended-Range Tanks

18. The two bladder-type main fuel cells (fig. 9) are located fore and aft of the ammunition bay. Each cell is self-sealing to 12.7mm projectiles with the 30-minute reserve portion of each cell self-sealing to 14.5mm projectiles. The forward cell capacity is 151 gallons (981.5 pounds) and the aft tank holds 202 gallons (1313 pounds). The cells are filled with reticulated flexible foam to suppress vapor ignition and flame propagation due to impact of high-explosive incendiary projectiles up to 23mm. They are also surrounded by a sprayed foam material to help keep the tanks from moving within their respective compartments and to prevent fuel vapors from forming in the voids between the fuel cells and the aircraft structure. Two lights on the caution panels illuminate when the fuel remaining reaches that required for 30 minutes of flight at maximum range airspeed at design gross weight.

19. Provisions are made for four externally mounted jettisonable fuel tanks. With the exception of the EXT TK switch and the EXT TANKS EMPTY caution light, the extended-range tank subsystem was not installed in the test aircraft.

Fuel Control Panels

20. Fuel control panels are provided in both crew stations. The master panel, located in the pilot crew station and illustrated in figure 4, contains switches for the following: fuel ON/OFF for each engine; cross-feed ON/OFF; boost pumps (BST PUMP) ON/OFF; external tank (EXT TK) ON/OFF; fuel transfer selector; and manual (MAN) fuel transfer selector. Two green advisory lights are provided to illuminate when the fuel lines are pressurized by the operating boost pump.

21. The emergency fuel (EMER FUEL) control panel, located in the copilot/gunner station and illustrated in figure 3, contains the following switches: BOOST PUMP ON/OFF; CROSSFEED ON/OFF; fuel transfer (TRANS) selector; and pilot override (PILOT OVERD) ON/OFF. With the pilot override switch in the OFF position, the system operates according to the switch positions selected on the pilot fuel control panel. Placing the override switch in the ON position overrides the pilot switches, enabling the copilot/gunner to assume control.

Vents

22. Each fuel cell incorporates a vent system that contains a valve which prevents fuel spillage in the event of aircraft rollover, and functions as a pressure relief valve to permit overflow in the event of failure of the pressure refueling system.

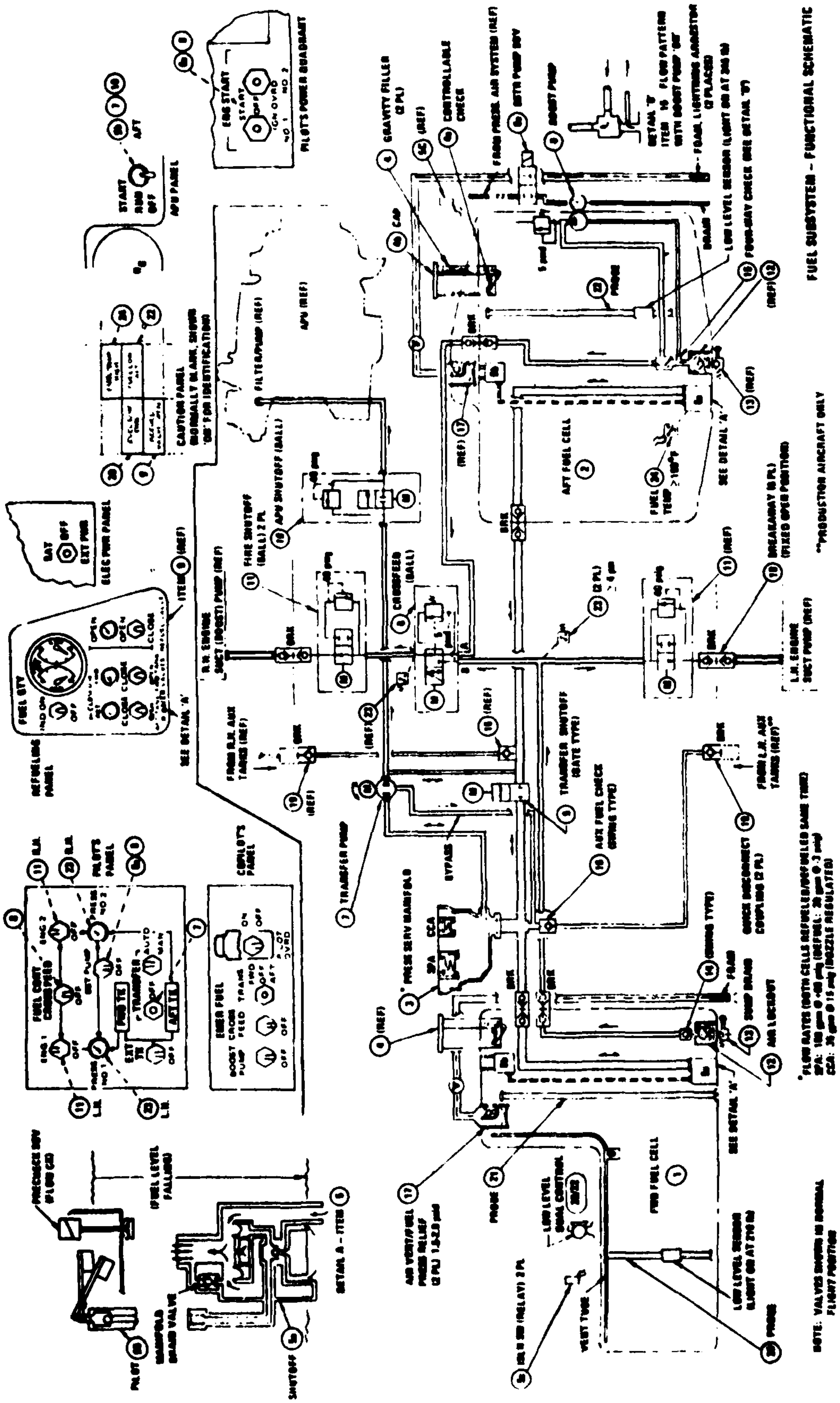


Figure 8. Fuel Subsystem Functional Schematic.

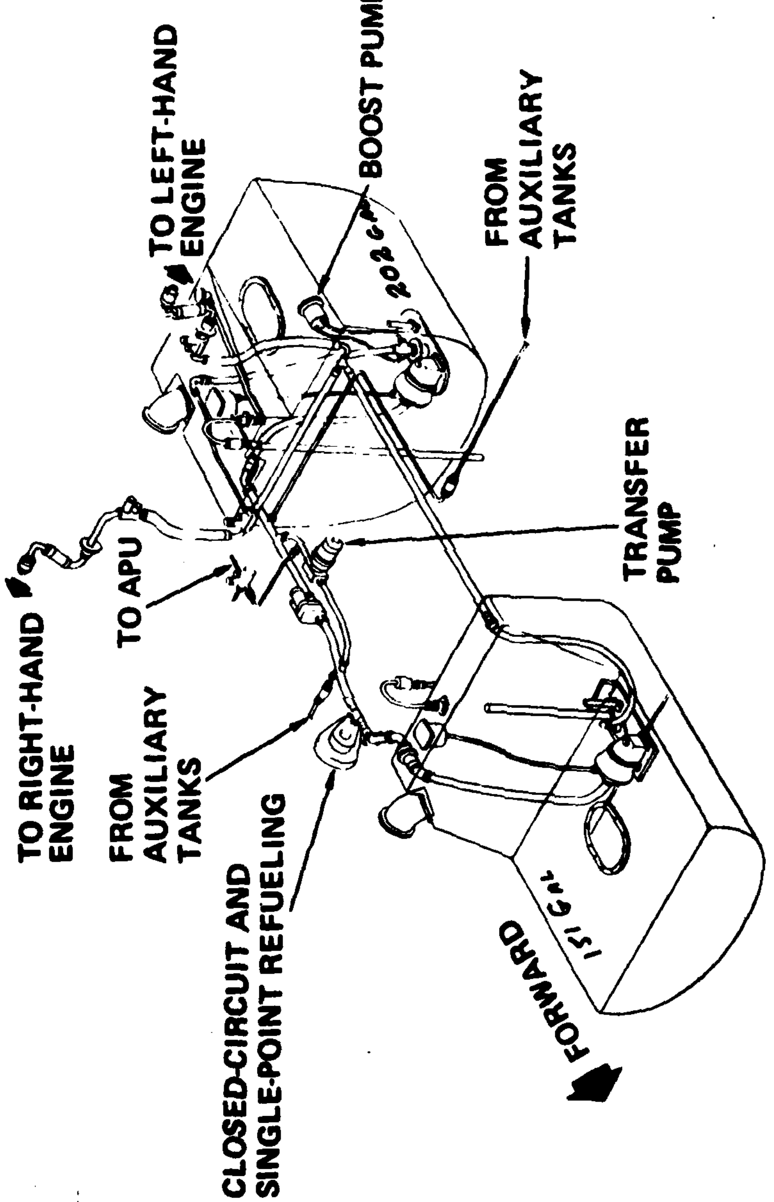


Figure 9. Fuel Subsystem.

Transfer Pump

23. A reversible-action electrically-driven pump permits transfer of fuel either way between fuel cells. Fuel transfer can be controlled from either crew station. In the pilot station, transfer is accomplished by placing the transfer switch in the desired forward or aft position. Fuel is transferred to the tank selected. The copilot/gunner can transfer fuel by placing the pilot override switch in the ON position and the selector switch to the desired tank. The transfer pump also provides fuel for APU starting and comes on automatically when the APU start switch is placed in the START position.

Cross-Feed

24. During normal operation, fuel is drawn by engine-mounted pumps from the forward fuel cell for the No. 1 (left) engine and from the aft cell for the No. 2 (right) engine. Valving is provided to allow both engines to be supplied fuel simultaneously from both tanks. The valve is turned on by placing the cross-feed valve switch in the ON position at either crew station. The copilot/gunner must place the pilot override switch in the ON position before his cross-feed switch is effective. The cross-feed system prevents engine flameout due to fuel starvation in the event of a fuel cell failure. Fuel level management is controlled by the fuel transfer system.

Boost Pump

25. An air-driven boost pump, located in the aft tank, is used during engine starting and at altitudes above 10,000 feet when fuel temperatures above 109°F (42.8°C) are encountered. A light on the caution panel illuminates when fuel temperature exceeds 109°F. The cross-feed valve opens automatically any time the boost pump is turned on. The boost pump comes on automatically when the engine start switches are activated. When needed, the pump can be turned on manually from either crew station. However, the copilot/gunner must turn the pilot override switch on in order to activate the boost pumps with his switch. Two green advisory lights on the pilot fuel control panel illuminate when fuel under boost pressure is being provided to the engines.

Refueling and Defueling

26. Each fuel cell has a gravity refueling receptacle. Refueling receptacles are also provided for closed circuit and single-point pressure refueling (photo 8). Lights on the caution panels at each crew station illuminate when the refueling valve is open. Refueling can be accomplished with the APU and/or the left engine (No. 1 engine) operating. Sump drains are provided in each main fuel cell. Gravity defueling may be accomplished by using these drains. Defueling can also be accomplished through the closed-circuit and single-point receptacles by using a fuel truck equipped with suction capability.

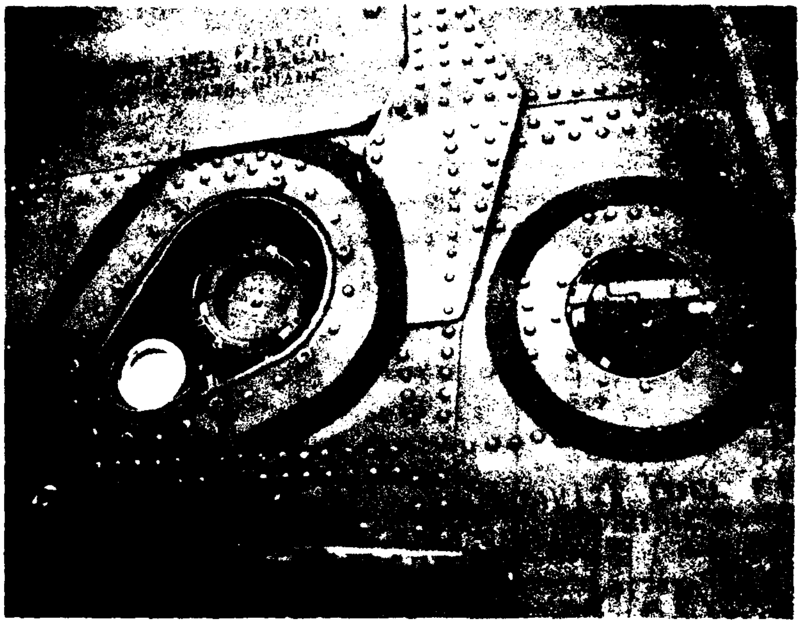


Photo 8. Refueling Receptacles.

ELECTRICAL POWER SYSTEM

27. The electrical power system, illustrated in figure 10, consists of all the components required for generation, conversion, distribution, and control of the aircraft AC and DC subsystems, the emergency electrical power subsystem, and interior/exterior lighting.

Primary Electrical Power Subsystem

28. The primary electrical power subsystem is of nominal 115/200-volt three-phase, 400-Hz design. The primary sources of AC power are two 20-KVA AC generators (alternators) mounted on the transmission accessory gearbox. The accessory gearbox is driven by the main transmission when the rotor system is turning and by the APU, on the ground, when the rotor system is not turning. To protect against underfrequency damage, both AC generators switch off-line after a 10-second delay whenever N_R is less than or equal to 90 percent and the APU is OFF. During normal operation, each AC generator supplies power to its own independent essential bus subsystem. A basic interlock bus configuration is used, which makes it physically impossible to parallel the two generators with each other or with external power. Aircraft electrical loads are distributed between the two

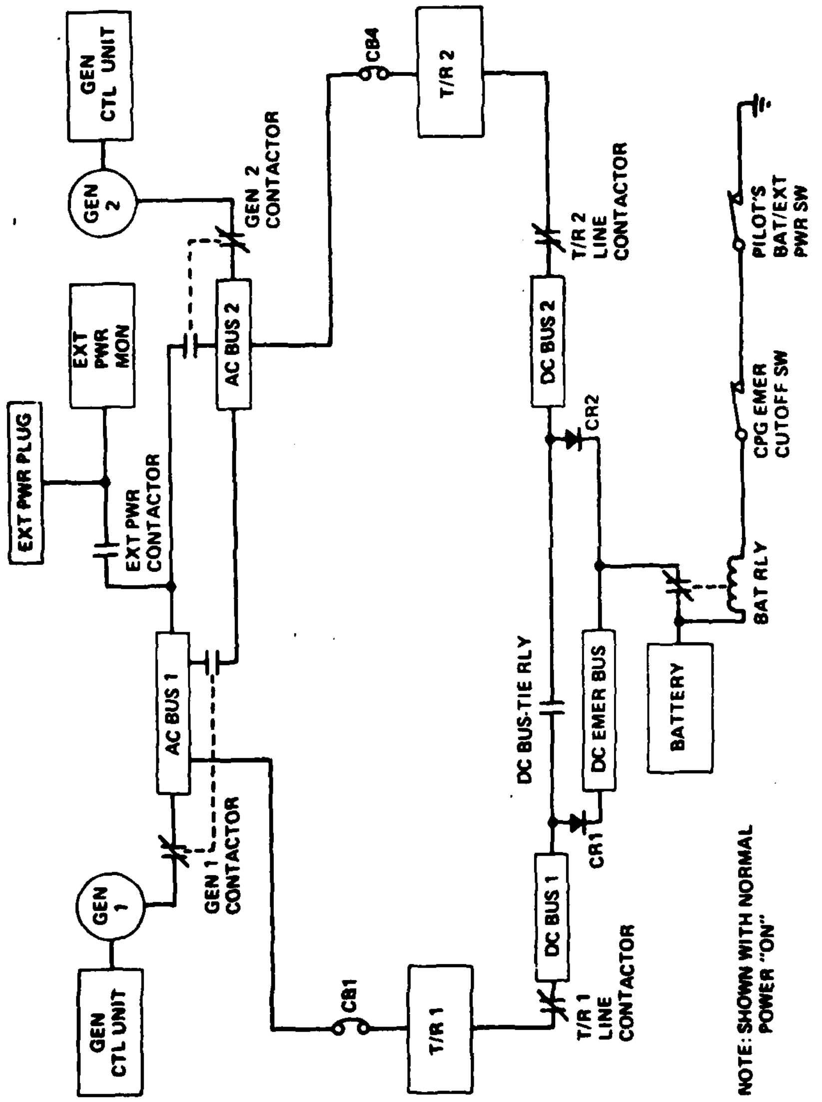


Figure 10. Electrical Power System.

AC buses, and each generator supplies approximately one-half of the total power requirement. Phase balance is considered when bus loads are assigned. If one generator fails to deliver power within specification limits, that generator is automatically disconnected from its bus and the bus is automatically connected in parallel with the other AC bus through the bus-tie relay; thus, the remaining generator provides electrical power for the aircraft. Each generator is controlled by its own generator control unit and is protected from overvoltage and undervoltage, underfrequency, and feeder faults. Each generator has adequate capacity (20 KVA) to supply the full aircraft electrical load.

Direct Current Electrical Power Subsystem

29. Direct current electrical power is nominally 28 volts, supplied by two 200-ampere transformers/rectifiers which are energized by AC electrical power. In normal operation, each transformer/rectifier supplies approximately one-half of the total DC aircraft electrical power requirement. If one transformer/rectifier fails it is automatically disconnected from its bus and the bus is automatically connected in parallel with the other DC bus through the bus-tie relay. The remaining operable transformer/rectifier furnishes all DC electrical power for the aircraft. In the event of loss of both tranformers/rectifiers, emergency DC bus loads are automatically fed by the battery and all DC loads except emergency loads are automatically switched off. Each transformer/rectifier has adequate capacity (200 amperes) to furnish all aircraft DC electrical power.

Emergency Direct Current Electrical Power

30. Emergency DC electrical power is furnished by a 12-cell, 17-ampere-hour, 5-hour-rate, 24-volt lead acid battery. The battery supplies electrical power to the aircraft DC emergency bus. The battery is connected directly to the DC emergency bus and receives charging current when that bus is energized with aircraft power. The battery is not on charge when the aircraft is on ground power. The battery also provides electrical power for entrance lighting, APU starting, maintenance lights, communication with ground crews (photo 9), and the tail wheel lock ground handling switch.

External Electrical Power

31. The 115/200-volt AC external electrical power can be utilized during ground operations by use of the external power receptacle (photo 10). The on-board external power monitor prevents external power from being applied to the aircraft if phase rotation, voltage, or frequency is out of specification. The device also monitors external power after it has been applied to protect aircraft electrical power-using equipment from out-of-specification ground power.

Photo 9. Ground Crew ICS Communication.

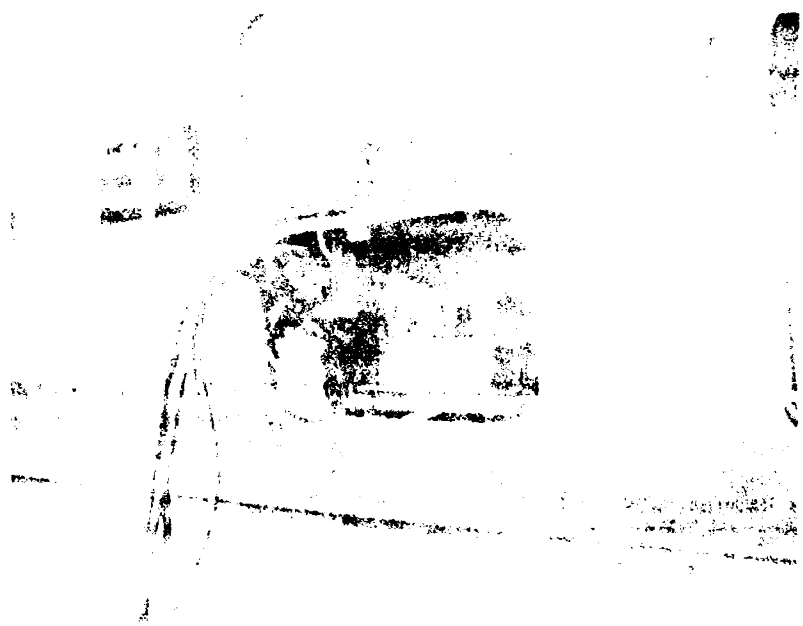


Photo 10. 115/200 Volt AC External Electrical Power Receptacle.

Aircraft Lighting

32. Exterior lighting includes navigation lights, anticollision lights, formation lights, a landing light, and a searchlight. Interior lighting includes flight and engine instrument lights, console panel lights, avionics panel lights, utility lights, and a walk-around inspection and maintenance light.

Crew Station Electrical Power Controls

33. An electrical power control panel, illustrated in figure 3, is provided in the pilot crew station. It contains two generator switches marked GEN NO. 1 and GEN NO. 2 with positions marked NORM, OFF/RESET, and TEST; an external power reset switch with positions marked BAT, OFF, and EXT PWR. With the battery/external power switch in the BAT or OFF positions the generators can be placed on-line any time the APU is running or N_R is greater than or equal to 94 percent. To put the generators on-line the switches are placed in the NORM position. In the event a generator should drop off-line during normal operation, as evidenced by a light on the caution panel, an attempt can be made to bring the generator back on-line by momentarily placing its control switch in the OFF/RESET position and then back to NORM. The TEST position for the generator switches is used to check generator circuits prior to placing the generators on-line.

34. The external power reset switch permits resetting the on-board external power monitor. If the monitor cannot be reset after two attempts, a malfunction of the external power source and/or the monitor has occurred.

35. A guarded battery switch on the copilot/gunner power quadrant (fig. 3) marked EMER BAT OVRD is normally in the NRML position. Placing the switch in the OVRD position overrides the pilot switch and disconnects the battery from the emergency bus.

36. The caution panels in both crew stations contain the following electrical power system lights: GEN 1 and GEN 2, which illuminate when the indicated AC generator is off-line; RECT 1 and RECT 2, which illuminate when the indicated transformer/rectifier is off-line; and TEMP HIGH RECT 1 and TEMP HIGH RECT 2, which illuminate when transformer/rectifier temperature becomes excessively high.

37. Two circuit breaker panels protecting the various AC and DC circuits are located in the pilot crew station, one on the left overhead canopy rail as illustrated in figure 11, and one on the inboard side of the left-hand console. A single panel is located on the copilot/gunner right-hand console. Circuit breaker nomenclature and the circuits which each circuit breaker protects are identified in tables 1, 2, and 3. In addition, one circuit breaker is located on the generator No. 2 power distribution box and four circuit breakers are located on the generator No. 1 power distribution box. These boxes are located on the stepway aft of the main

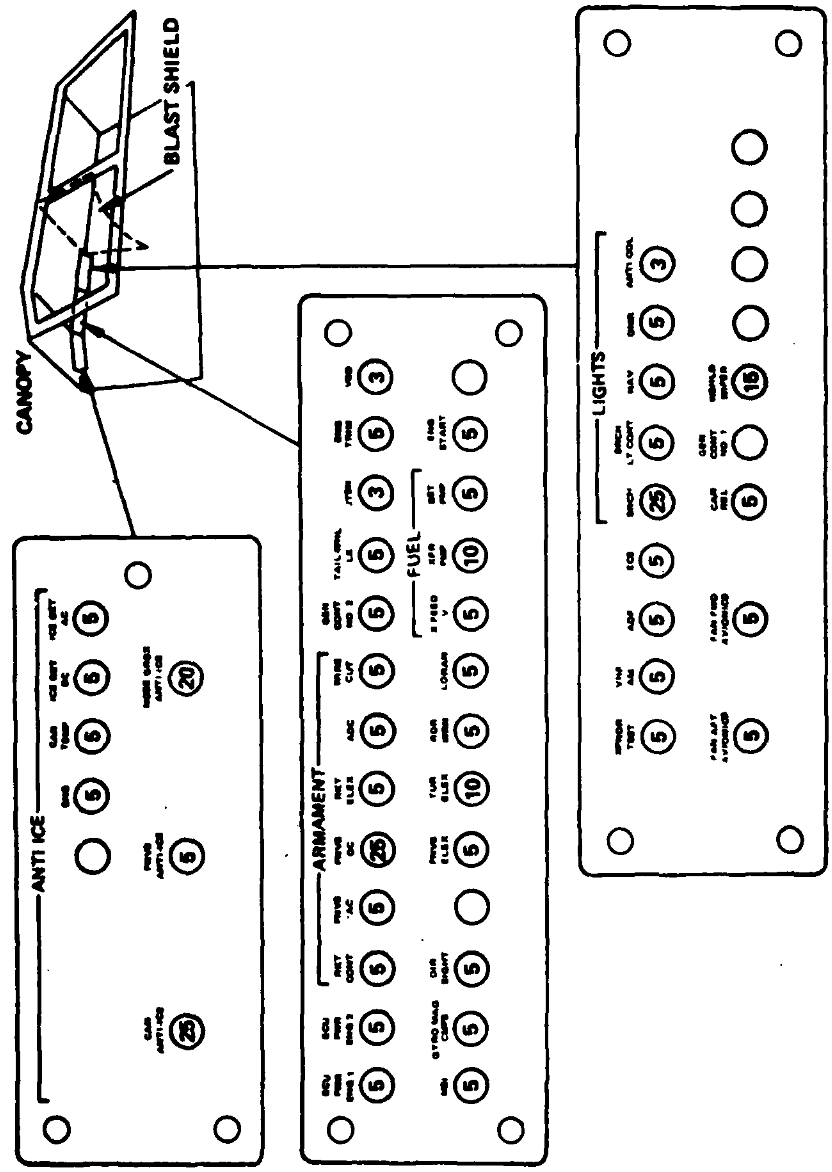


Figure 11. Pilot Canopy Rail Circuit Breaker Panels.

Table 1. Pilot Overhead Circuit Breakers.

Nomenclature		Circuits
Anti-ice	CAN ANTI-ICE	Canopy Heater
	PNVS ANTI-ICE	Inactive
	NOSE GRBX ANTI-ICE	Engine nose gearbox heaters
	ENG	Engine inlet anti-ice
	CAN TEMP	Total air temp sensor*
	ICE DET DC	Inactive
	ICE DET AC	Inactive
	FCU PWR ENG 1	Engine 1 control unit auxiliary and backup power
	FCU PWR ENG 2	Engine 2 control unit auxiliary and backup power
Armament	RKT CONT	Inactive
	PNVS AC	Aircraft 74-22248 Inactive Aircraft 74-22249 radar altimeter
	PNVS DC	Inactive
	RKT FLEX	Inactive
	ADC	Inactive
	WIRE CUT	Inactive
	GEN CONT NO. 2	Generator control unit no. 2
	TAIL-WHL LK	Tailwheel lock system (except tailweel lock bypass)
	JTSN	Pilot stores jettison controls
	ENG TRIM	Engine beep, engines cut circuit, transmission diverter valve
	XPNDR TEST	Inactive
	VHF AM	ARC-115 VHF-AM radio
	ADF	Inactive
	ECS	Environmental control system
	CAN REL	Canopy heater control, nose gearbox heater relay
	GEN CONT NO. 1	Generator control unit no. 1
	WSHLD WIPER	Windshield wipers

*Flight test installation

Table 2. Pilot Lower Left Console Circuit Breakers.

Nomenclature		Circuits
ASE	DC	ASE, BUCS release relays ASE computer
	AC	ASE computer, BUCS release relays
Gyro	STBY	Standby gyro
	VERT	Vertical gyro
Lights	CAUT	Caution panel
	LDG	Landing light
	UTIL	Floodlights, utility light
	LDG CONT	Landing light control
Radio	UHF	ARC-116 UHF-AM radio
	ICS	Intercommunications control set
	FIRE EXT	Pilot's engine fire extinguishers
	FIRE EXT	APU fire extinguisher
	BUCS	Backup control system (BUCS)
	TRIM	Trim release solenoids, squat switch, squat relay, indicated airspeed transducer
	XPNDR TR	Inactive
	RDR ALTM	Inactive
Fire Det	APU	APU fire detectors
	ENG 1	Engine 1 fire detectors
	ENG 2	Engine 2 fire detectors
	FIRE EXT	Copilot/gunner's fire extinguishers
Fuel	MAIN ACTR	Main fuel valve actuators
	XFR PMP	Transfer pump
	FILL	Refueling controls
	MGT	Fuel management
	LVR	Engine louver actuators
	APU	APU, shaft-driven compressor throttle valve, APU fuel firewall valve
	JTSN	Copilot/gunner stores jettison control
	EMER HYD	Emergency hydraulic switch, R&L throttle release solenoid
	PITOT HT	Pitot heat
	INSTR PRIM	Pilot engine instruments, hydraulic pressure transducers
	RTR BK	Rotor brake system, ASE self test switch

Table 3. Copilot/Gunner Circuit Breakers.

Nomenclature	Circuits
LASER	Inactive
DIR SIGHT	Inactive
ELEV DR	Inactive
ELEV DR	Inactive
VHF FM	Inactive
INST PRIM	Copilot/gunner engine instruments
HELM SIGHT	Inactive
ANTI-ICE	Inactive
CAUT LT	Caution light panel
UTIL LT	Utility light, floodlights, advisory light dimmer
ALTM VIB	Pilot and copilot/gunner altimeter vibrator
LASER PWR	Inactive
ORD DR MTR	30mm gun drive motor
NVS AND LASER WINDOW	Inactive
TOW PWR SPLY	Inactive
NVS	Inactive
DMR LT	Light dimmer
TME	Inactive
TOW SQUIB	Inactive
NVS GYRO	Inactive
NVS	Inactive
DIR SIGHT	Inactive
FORM LT	Formation lights
FCC	Inactive

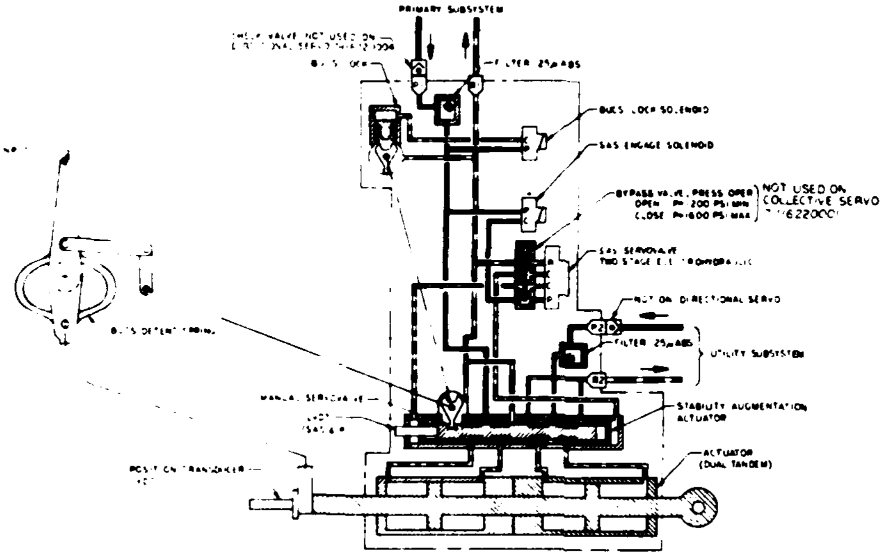
transmission. All circuit breakers are of the resettable push/pull type. Any malfunctioning circuit may be isolated from its power supply system by pulling out its circuit breaker.

38. A dual-scaled DC ammeter and a dual-scaled AC load meter are installed in the pilot instrument panel. These indicators display the load being carried by each AC generator and transformer/rectifier. As each AC generator and transformer/rectifier has the capacity to supply the entire AC or DC load, the ammeter and load meter are used to confirm the existence of malfunctions.

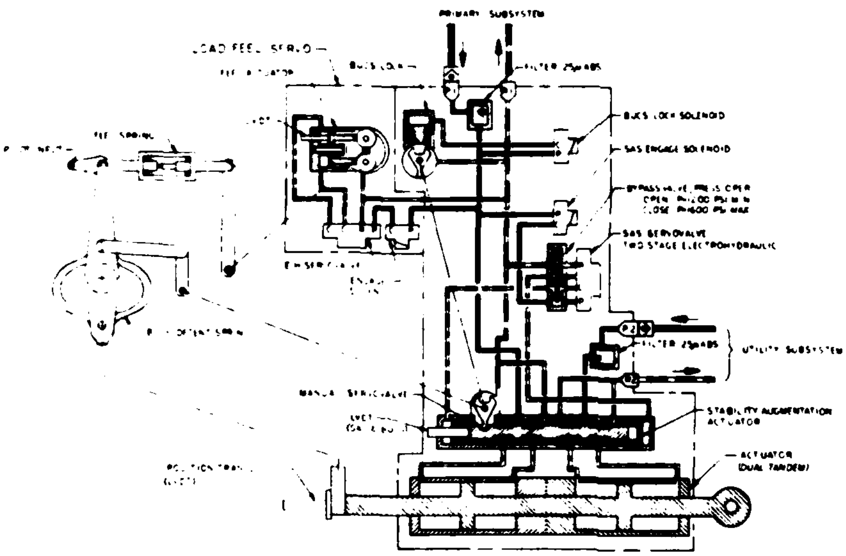
HYDRAULIC SYSTEM

39. Two independent hydraulic systems are provided. The primary system provides power for the cyclic, collective, and tail rotor controls only. The utility system provides redundant power for the flight controls as well as powering the wing flaps, rotor brake, area weapon aiming, external stores elevation, and APU starting. A failure of one system will not adversely affect the other system. Each system is powered by a separate hydraulic pump that operates at 3000 psi. The pumps, driven off the main transmission accessory gear train, operate whenever the main rotor is turning to ensure hydraulic pressure during autorotation. The pumps are also driven off the accessory gear train by the APU, on the ground, with the rotor systems stopped and engines shut down. In addition to the pressure pumps, each hydraulic system contains the following: manifold containing a reservoir, a switch to provide warning of low reservoir fluid level, relief valve, pressure transducer for the hydraulic pressure gauge on the pilot instrument panel, a pressure switch to provide warning of pump failure, and a filter with bypass valve; heat exchanger to maintain fluid below 275°F (135°C); and an accumulator. The utility manifold also contains the rotor brake controls and isolation valves. Major system components are illustrated in figure 12.

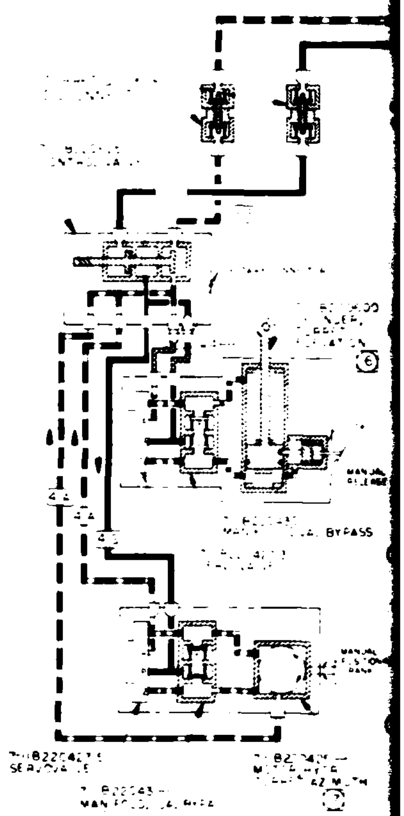
40. The reservoirs are pressurized by compressed air from the pressurized air system manifold. The accumulators are precharged with nitrogen gas. Fluid level gauges are provided on the reservoirs and gas pressure gauges are provided on the accumulator. When utility accumulator oil pressure is discharged the gas pressure gauge on the accumulator will read near 1600 psi. A dual reading gauge on the pilot instrument panel (fig. 4) indicates hydraulic pressure in the primary system and in the utility accumulator. A failure in the utility pump does not always result in a drop in utility accumulator pressure. Lights are provided on the caution panels in both crew stations to signify actual or impending failures. Emergency hydraulic switches are provided in both crew stations. When the emergency switch is in the ON position, the utility accumulator provides power to the flight control servo actuators only. All other functions are isolated from the system. The aircraft is designed to remain controllable and safe to fly after failure of either the primary or utility system. In the event of a utility system failure after a primary system failure, the utility accumulator is designed to supply adequate hydraulic power to make an immediate landing. The auxiliary functions of the utility hydraulic

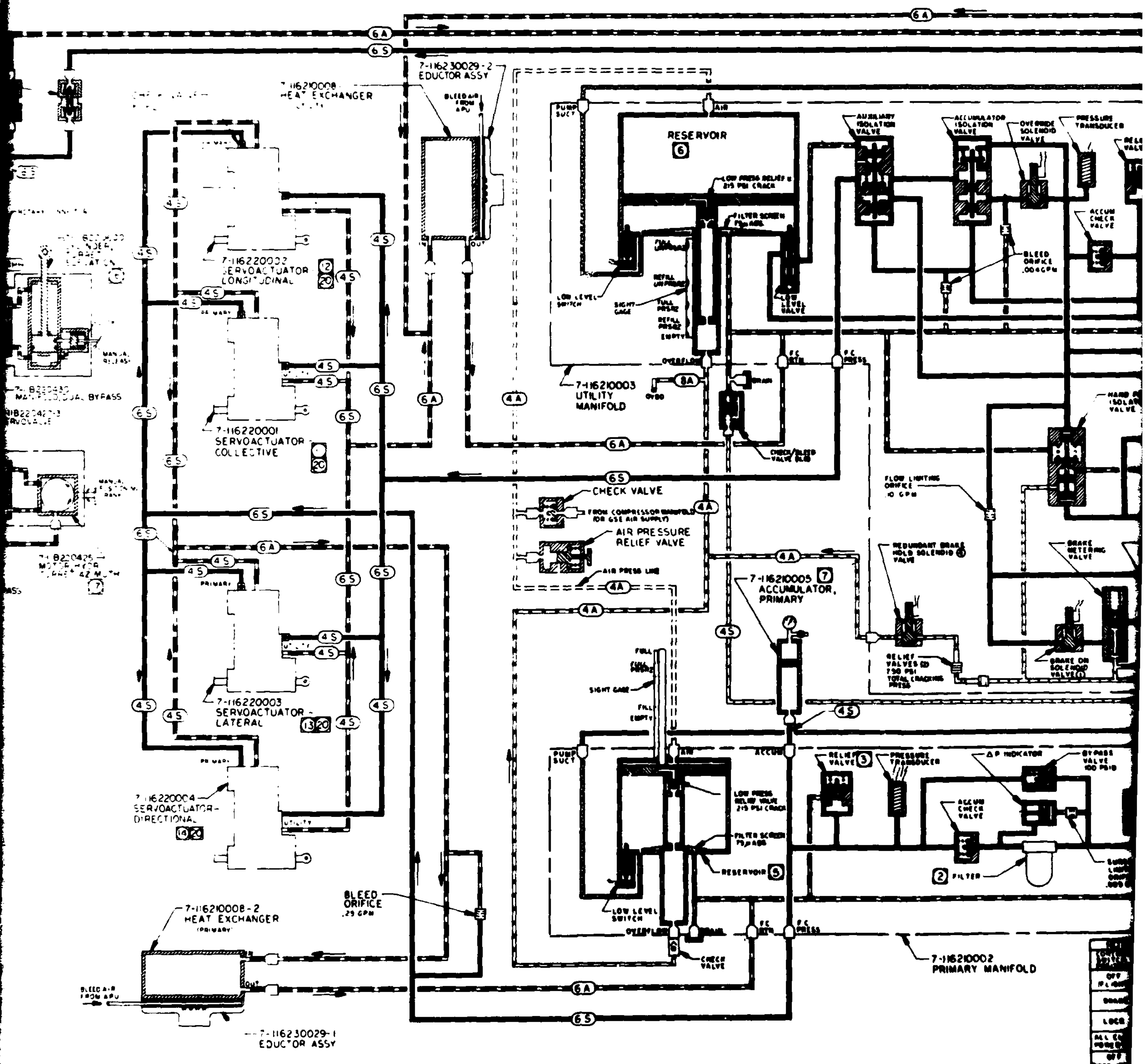


SERVOACTUATOR
7-116220001 COLLECTIVE (11) (EXCEPT AS NOTED)
7-116220003 LATERAL (13)
7-116220004 DIRECTIONAL (14)



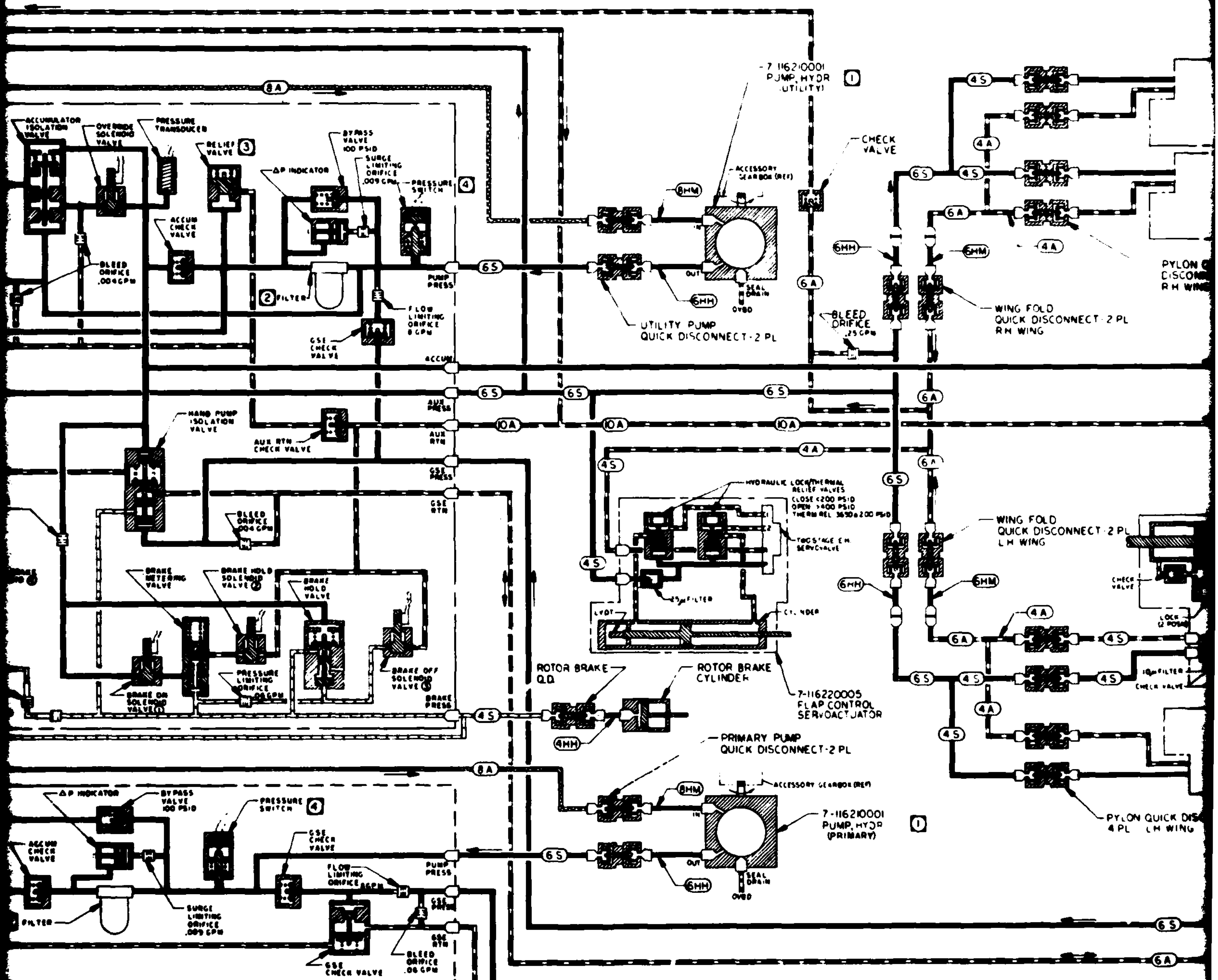
7-116220002 LONGITUDINAL SERVOACTUATOR (12)




6A
6S
4S
4A
8A
7-116230029-2
EDUCTOR ASSY
7-116210008
HEAT EXCHANGER
RESERVOIR
6
7-116220002
SERVOACTUATOR
LONGITUDINAL
7-116220001
SERVOACTUATOR
COLLECTIVE
7-116210003
UTILITY
MANIFOLD
CHECK VALVE
AIR PRESSURE
RELIEF VALVE
7-116210005
7
ACCUMULATOR,
PRIMARY
7-116220003
SERVOACTUATOR
LATERAL
7-116220004
SERVOACTUATOR
DIRECTIONAL
BLEED
ORIFICE
7-116210008-2
HEAT EXCHANGER
7-116230029-1
EDUCTOR ASSY
7-116210002
PRIMARY MANIFOLD
RESERVOIR 5
2 FILTER
RELIEF VALVE 3
PRESSURE TRANSDUCER
ΔP INDICATOR
BYPASS VALVE 100 PSID
ACCUM CHECK VALVE
LOW LEVEL SWITCH
CHECK VALVE
SIGHT GAGE
FULL
EMPTY
PUMP SUCT
AIR
OVERFLOW
DRAIN
FC RTN
FC PRESS
ACCUM
FLOW LIMITING ORIFICE .10 GPM
BLEED ORIFICE .004 GPM
ACCUM CHECK VALVE
AUXILIARY ISOLATION VALVE
ACCUMULATOR ISOLATION VALVE
OVERRIDE SOLENOID VALVE
PRESSURE TRANSDUCER
BRAKE METERING VALVE
REDUNDANT BRAKE HOLD SOLENOID VALVE
RELIEF VALVES 750 PSI TOTAL CRACKING PRESS
BRAKE ON SOLENOID VALVE
CHECK/BLEED VALVE
AIR PRESS LINE
MANUAL RELEASE
BYPASS

.2

1



116210002
PRIMARY MANIFOLD

ROTOR BRAKE OPERATION SEQUENCE		
CONTROL SWITCH POSITION	SOLENOID VALVE OPERATION	ROTOR BRAKE FUNCTION
OFF (FLIGHT)	OFF ON	BRAKE PRESS = RETURN PRESS
BRAKE	OFF ON	METERING VALVE REGULATES BRAKE PRESSURE TO 590 PSI
LOCK	OFF ON	BRAKE PRESS = ACCUM PRESS BRAKE HOLD VALVE OPEN
ALL ELEC POWER OFF	ALL OFF	BRAKE REMAINS ON NO LEAKAGE TO RETURN
OFF (PREFLIGHT)	OFF ON	BRAKE PRESS = RETURN PRESS

13

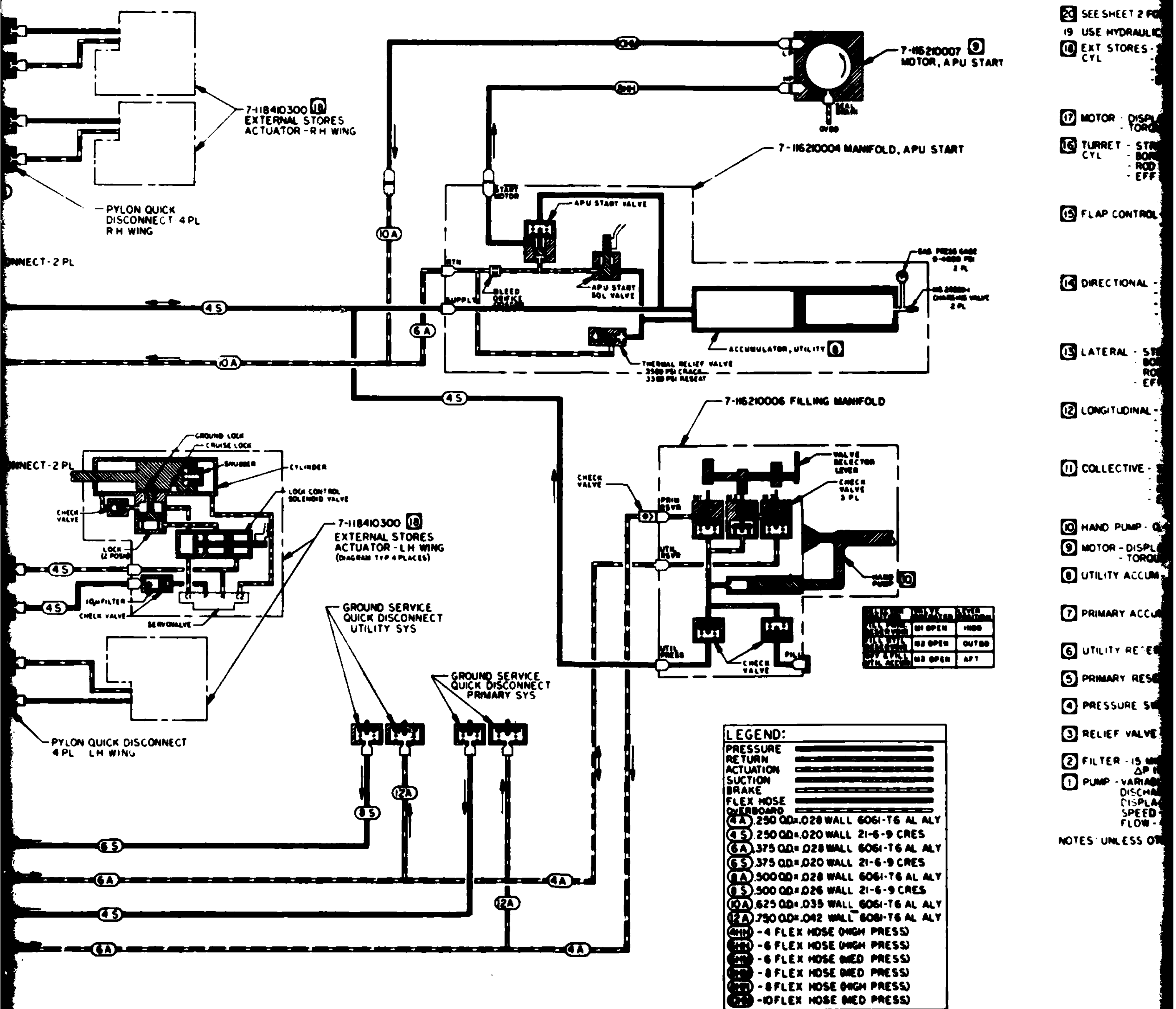
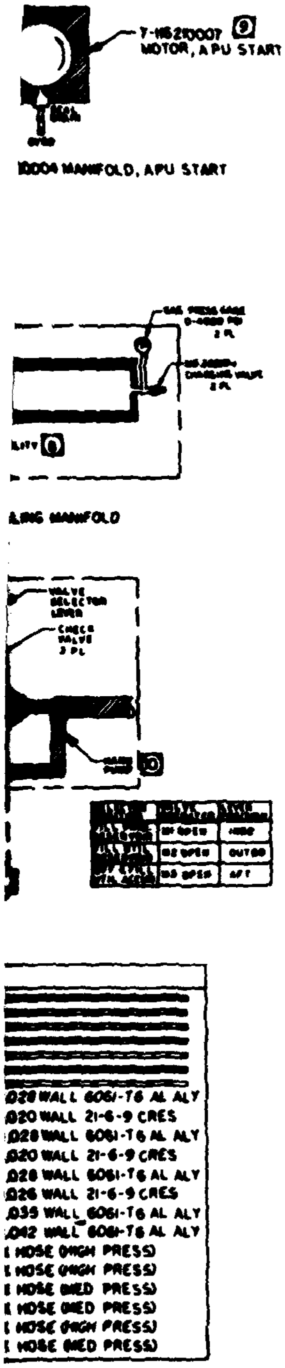


Figure 12. Hydr

FOR OFFICIAL USE ONLY

14



20 SEE SHEET 2 FOR ACTUATOR DETAIL

19 USE HYDRAULIC FLUID PER MIL-H-5606

18 EXT STORES CYL - STROKE 5.75 IN.
- BORE DIA 1.25 IN
- ROD DIA .68 IN
- EFF AREA EXT 1.22 IN²
RETR .86 IN²

17 MOTOR - DISPLACEMENT .095 CIPR
- TORQUE AT 3000 PSI - 21.9 IN LB

16 TURRET CYL - STROKE 6.00 IN
- BORE DIA 1.125 IN
- ROD DIA .625 IN
- EFF AREA EXT .985 IN²
RETR .678 IN²

15 FLAP CONTROL - STROKE 4.97 IN
- BORE DIA .750 IN
- ROD DIA .622 IN, .584 IN
- EFF AREA .1738 IN² RETR
.1378 IN² EXT

14 DIRECTIONAL - STROKE ±1.750 IN
- BORE DIA 1.428 IN, 1.215 IN
- ROD DIA 1.372 IN, 1.149 IN, 0.872 IN
- EFF AREA 1.23 IN² EXT/SYS
.563 IN² RET/SYS

13 LATERAL - STROKE ±1.750 IN
- BORE DIA 1.179 IN
ROD DIA .934 IN
- EFF AREA .407 IN²/SYS

12 LONGITUDINAL - STROKE ±1.750 IN
- BORE DIA 1.679 IN
- ROD DIA 1.069 IN
- EFF AREA 1.333 IN²/SYS

11 COLLECTIVE - STROKE 1.750 IN EXT; 1.551 RETR
BORE DIA 1.871 IN
ROD DIA 1.05[illegible] IN
- EFF AREA 1.869 IN²/SYS

10 HAND PUMP - 0.4 IN³/ STROKE

9 MOTOR - DISPLACEMENT 0.6 CIPR
- TORQUE AT 3000 PSI 286 IN LB

8 UTILITY ACCUM - GAS PRECHARGE 1640 PSIG AT 70°F
TOTAL VOLUME - 287.3 IN³
- FLUID VOLUME - 162.6 IN³

7 PRIMARY ACCUM - AIR PRECHARGE - 1640 P[illegible] AT 70°F
- TOTAL VOLUME - 8.05 IN³
FLUID VOLUME - 4.8 IN³

6 UTILITY RESERVOIR - FLUID VOLUME 270 IN³
- AIR PRESS 20 - 68 PSIG

5 PRIMARY RESERVOIR FLUID VOLUME 30 IN³
AIR PRE[illegible] 20 - 68 PSIG

4 PRESSURE SWITCH OPEN ABOVE 2450 PSIG
- CLOSED BELOW 1500 PSIG

3 RELIEF VALVE - CRACKING PRESS 3500 PSIG
RESEAT PRESS 3300 PSIG

2 FILTER - 15 MICRON ABSOLUTE
ΔP IND POPOUT AT 70±10 PSID, >100±15°F

1 PUMP - VARIABLE DISPLACEMENT
DISCHARGE PRESSURE - 3000 PSIG
DISPLACEMENT - .085 CIPR
SPEED - 12,635 RPM
FLOW - 4.1 GPM

NOTES: UNLESS OTHERWISE SPECIFIED

Figure 12. Hydraulic Subsystem Schematic.

IAL USE ONLY

5

system are isolated from the flight control portion of the system in the event of a low fluid level condition in the utility reservoir. If the utility pump fails and pressure is lost, isolation of the utility accumulator pressure is automatically sequenced. In this way the utility accumulator is usable as a reserve source of flight control servo actuator power.

LANDING GEAR SYSTEM

41. The aircraft is equipped with fixed landing gear consisting of two main wheels and a tail wheel. The gear provides stability for taxi, takeoff, and landing at all gross weights on terrain with slopes of up to 12 degrees and for landing sideways on a 15-degree slope under zero wind conditions. The main and tail gear incorporate air-oil shock absorbers. Hydraulic brakes on the main wheels are manually controlled by brake pedals in both crew stations. A parking brake lock lever is located in the aft crew station. The tail wheel is 360 degrees free-swiveling and self-centering. A tail wheel lock is electrically operated by two switches. One switch is located on the tail wheel control panel in the pilot left console, as illustrated in figure 4. Adjacent to the switch is an advisory light which illuminates when the tail wheel is unlocked. The second is located on the aircraft right rear electrical service panel. The main gear incorporates a ground crew-operated kneeling capability to facilitate air transport. A sensor is provided to signal when the aircraft weight is on its wheels. Its location is on the left main gear trailing strut. The sensor's function is to enable aircraft ordnance systems to be activated when airborne.

INSTRUMENTS

42. The aircraft is fitted with standard and specialized flight instruments; vertical tape engine instruments (aircraft SN 74-22248) or Canadian Marconi vertical scale digital fiberoptic instruments (aircraft SN 74-22249); electrical power meters; hydraulic system pressure instruments; and fuel quantity instruments, as seen in figures 3 and 4.

Flight Instruments

43. A basic set of standard flight instruments is provided for each crew station. Both the pilot and the copilot/gunner are presented with airspeed indicator, barometric altimeter, instantaneous vertical speed indicator, and attitude indicator. The pilot attitude indicator is a standby unit. It is completely self-contained and operates from the emergency bus. Copilot/gunner attitude indicator is AC-powered and slaved to the aircraft vertical gyro. In addition, the pilot is supplied with magnetic compass, outside air temperature (OAT), accelerometer, clock, VSD, and turn-and-slip indicator. With the exception of the OAT gauge and the magnetic compass, all flight instruments are mounted in the pilot instrument panel and in the copilot/gunner right instrument panel. A VSD in the pilot instrument panel presents aircraft pitch and roll attitude.

Engine Instruments

44. Engine parameters in aircraft SN 74-22248 are displayed by vertical scale instruments which present a vertical-moving tape with fixed scale. Engine instrument faces are illustrated in figures 3 and 4. Normal ranges for all parameters are located toward the center of each scale; therefore, abnormally high or low readings stand out from centered normal readings. Scales are marked to permit a high degree of quantitative accuracy. Standardized color codes are utilized. Red "tick" marks indicate red-line values which should not be exceeded during normal operations. Amber "hatched" zones indicate marginal or transitional ranges. Solid green zones indicate normal operating ranges. Engine operation within these ranges is unlimited. Under red night illumination, these colors "wash out," and color codes become useless. Therefore, shape codes are incorporated into the color codes to facilitate night interpretation. Red line values show up as solid, uniquely shaped "tick" marks, amber marginal zones become "hatched" areas, and green normal zones become thickened solid areas.

45. Engine parameters in aircraft SN 74-22249 are displayed by vertical scale instruments which present an ascending chain of lighted segments with a fixed scale. The pilot is provided with a three-scale N_P and N_R indicator and two-scale indicators for torque turbine gas temperature, N_G, engine oil temperature, and engine oil pressure, as illustrated in figure 4. The torque, turbine gas temperature, and N_G indicators include an integral digital display at the bottom of each scale, providing accurate quantitative data on engine parameters at all times. A separate panel-mounted device is provided to enable the pilot to activate the instrument test feature. A control is also provided for dimming the display segments and digital displays. Scale lighting is controlled by the ENG T dimmer on the INT LT control panel. The copilot/gunner is provided with a three-scale N_P and N_R indicator, two-scale torque and fuel quantity indicators, and a selectable display device, all of which are illustrated in figure 3.

46. The selectable display device includes two digital displays, six display identifying labels with an indicator lamp next to each, a display selector rotary switch, instrument test enable button, and a dimmer to control the brightness of the display segments and the digital displays. Brightness of the scales is controlled by the ENG INST dimmer on the INT LT control panel. The copilot/gunner may use the selectable display device to obtain quantitative data on five different parameters: fuel quantity in both fuel cells, engine turbine gas temperature, N_G, engine oil pressure, and engine oil temperature. The sixth channel is a spare. To call up any one of these five quantities, the copilot/gunner turns the rotary switch labeled SELECT. The switch has six detented positions. When moved to any position a signal light appears next to the selected parameter and the current value of the parameter is digitally displayed. All vertical instruments in aircraft SN 74-22249 are read by noting the position of the uppermost lighted segment with respect to the fixed scale. These segments light, and extinguish vertically, a segment of time. Each lighted segment is colored to reflect the operating zone in which it resides. Segments in normal operating zones are green, those in

cautionary zones are yellow, and those at upper red-line values and above are red. The instruments are powered by redundant power supplies. Failure of either power supply results in the following:

a. The AUX PWR indicator on the vertical scale instrument control in the pilot crew station switches ON.

b. Every other segment in all vertical scales powered by the failed power supply and the associated digital displays, where applicable, switch OFF.

47. Figure 13 illustrates the results of a failure in either power supply. Only the torque indicator is presented. In actuality all vertical scale instruments would present similar displays. To eliminate the possible confusion caused by presenting yellow and/or red-lighted segments below green-lighted normal zones, a unique feature is employed. As the value of a parameter increases to the point at which the first green segment is activated, all lower yellow and/or red segments switch off. This logic is reversed as the value of the parameter decreases. For example, N_R values less than 94 percent are considered below red line. Therefore, all N_R segments from zero to 93 percent are colored red. After engine start, as rotor speed increases, red segments will light up to display N_R. When N_R is 93 percent, all lower, and hence red-colored segments, will be displayed. When N_R becomes 94 percent, the first green segment illuminates and all lower red segments switch off. The procedure works in reverse as N_R decreases below 94 percent. Table 4 presents the vertical scale instrument color scheme.

<u>Electrical Meters</u>

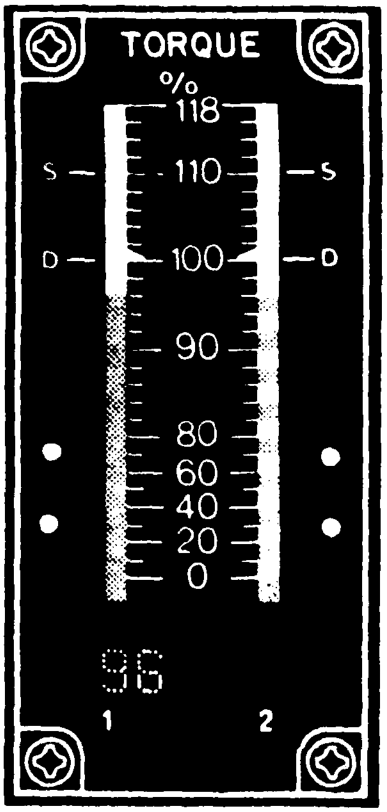
48. Aircraft AC and DC electrical loads are displayed to the pilot by a set of two dual-scaled AC load meters and DC ammeters. Their dials are illustrated in figure 4.

<u>Hydraulic Pressure Instrument</u>

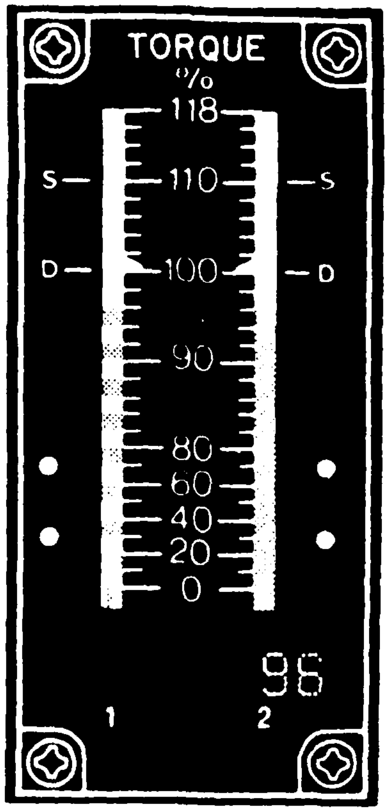
49. The pilot is presented with a display of primary hydraulic system and utility hydraulic accumulator pressure via a dual-scaled instrument. This instrument is illustrated in figure 4.

<u>Fuel Quantity Instruments</u>

50. Fuel quantity instruments in aircraft SN 74-22248 are of the same type as the engine instruments - vertical moving tape with fixed scale. The pilot is provided with a triple-scale fuel quantity indicator. Separate tapes present fuel quantity information for aft and forward fuel cells, while total fuel quantity is displayed on the third tape. The copilot/gunner is provided with a dual-scale indicator displaying only forward and aft cell quantity. The gauges are checked at each crew station by pressing the engine instrument preflight test switch. The gauges should read half-scale when the switch is depressed. Both instruments are illustrated in figures 3 and 4.



No. 1 Power
Supply Failure



No. 2 Power
Supply Failure

Pilot's AUX PWR
Indicator ON

NOTE: Both engines delivering 96 percent torque each.

Figure 13. Vertical Scale Instrument Power Supply Failure.

Table 4. Vertical Scale Instrument Color Scheme.

Scale	Red Zone(s)	Amber Zone(s)	Green Zone
N_P	0 to 98% 100 to top		98 to 100%
N_R	0 to 94% 104 to top		94 to 104%
TORQUE	100 to top	Amber dot at 30% Amber dot at 70%	0 to 100%
TGT	850 to top Red dot at 875°	775° to 850°	0° to 775°
N_G	0 to 65% 105 to top	102 to 105%	65 to 102%
FUEL QTY		FWD: 0 to 200 lb AFT: 0 to 250 lb	FWD: 200 to top AFT: 250 to top
ENG OIL PRES	0 to 25 psi 100 to top		25 to 100 psi
ENG OIL TEMP	150 to top	-60° to 30°	30° to 150°

51. The fuel quantity instruments in aircraft SN 74-22249 are of the same type as the engine instruments - an ascending chain of lighted segments with a fixed scale. The pilot indicator includes digital displays of fuel quantity and is illustrated in figure 4. The copilot/gunner receives quantitative fuel quantity information by selecting FUEL QTY on his selectable display. The selectable display is illustrated as part of figure 3.

CAUTION, WARNING, AND ADVISORY SYSTEM

52. Caution, warning, and advisory light signals are provided. Additionally, an auditory tone signal is provided for engine-out and low rotor speed warning.

Master Caution/Warning Panels

53. Master caution/warning panels are located at the top center of the pilot and the top right of the copilot/gunner instrument panels. Both crew stations contain identical master caution/warning panel capabilities. Their configurations are illustrated in figures 3 and 4 and their master caution/warning functions are described in tables 5 and 6. The master caution and both engine-out warning devices are combination switch lights. The master caution is reset by pressing the MASTER CAUTION switch light, while the engine-out auditory signal is reset by pressing the appropriate ENG 1 OUT or ENG 2 OUT switch light. Lighted fire handles warn the crew when fire is detected in either engine or in the APU. The location and function of each handle is described in table 7. Lamp testing of all caution, warning, and advisory lights in either crew station is accomplished by pressing the PRESS TO TEST switch.

Caution Panels

54. A caution panel is located in the lower right portion of the pilot instrument panel as well as in the copilot/gunner right console. The caution panel configuration for both crew stations is illustrated in figures 3 and 4. Table 8 functionally describes the operation of each caution channel and suggests the appropriate crew response to each.

Advisory Lights

55. A limited number of advisory lights are colocated with the equipment controls with which they are related. The advisory lights are identified and functionally described in table 9.

Table 5. Pilot and Copilot/Gunner Warning Panel Devices.

DEVICE LABEL	TYPE OF DEVICE	LIGHT COMES ON WHEN	LIGHT GOES OFF WHEN	NOTES
ROTOR RPM LO	Light	Rotor speed N_R drops below 90.3%.	Rotor speed increases above 90.3% N_R.	
ROTOR RPM HI	Light	Rotor speed N_R rises above 104%.	Rotor speed decreases below 104%.	
ENG 1, 2 OUT	Light/switch	N_G of indicated engine drops below 67% -OR- if engine power lever is advanced to FLY, N_P, of indicated engine drops below 96%.	N_G rises above 67% and, if engine power lever is advanced to FLY, N_P rises above 96% on indicated engine.	(1) (2)
MASTER CAUTION	Light/switch	Any caution light switches ON.	Manually reset -OR- the last caution light goes off.	(3)
APU FIRE	Light	APU fire is detected.	APU fire is extinguished.	
ENGS CHOP	Light	Engine power chopped to idle range by pressing ENG CUT button.	Either crewmember deactivates power chop feature by pressing ENG CUT button a second time.	
PRESS TO TEST	Switch			(4)

NOTES: (1) Press and release to reset Engine Out Auditory Warning Signal.
(2) N_P engine out warning is inoperative if the engine power lever is retarded to IDLE.
(3) Press and release to reset.
(4) Pressed and held to test all caution, warning, and advisory lamps.

Table 6. Aircraft Engine-Out Warning Systems.[1]

System	Activated By	Pilot Presentation
Engine	$N_P \leqq 96\%$ or $N_G \leqq 67\%$	Aural tone and engine-out light
Rotor	$N_R \leqq 90.3\%$	Aural tone and rotor speed low

[1]With PCL in flight-idle position, N_P system is deactivated.

Table 7. Pilot and Copilot/Gunner Fire Handles.

HANDLE LABEL	HANDLE LOCATION	LIGHT COMES ON WHEN	LIGHT GOES OFF WHEN	NOTES
ENG FIRE PULL 1	Upper left instrument panel	Engine 1 fire is detected	Engine 1 fire is extinguished	
ENG FIRE PULL 2	Upper left instrument panel	Engine 2 fire is detected	Engine 2 fire is extinguished	
APU FIRE PULL	Pilot's right console only	APU fire is detected	APU fire is extinguished	(1)

NOTES: (1) APU FIRE warning light and light in fire warning handle switch ON simultaneously.

Table 8. Pilot and Copilot/Gunner Caution Panel Lights.

CAUTION LABEL	MEANING	CREW RESPONSE	LIGHT GOES OFF WHEN	NOTES
OIL HOT ENG 1, 2	Engine oil temperature is high	Shut down engine or reduce power LAND AS SOON AS PRACTICAL	Oil temperature is less than critical value	
OIL PRESS ENG 1, 2	Engine oil pressure is low	Shut down engine or reduce power LAND AS SOON AS PRACTICAL	Oil pressure is greater than critical value	
OIL FLTR BYP ENG 1, 2	Engine oil is not being filtered	Shut down engine or reduce power LAND AS SOON AS PRACTICAL	Oil filter is operating normally	(1)
FUEL FLTR ENG 1, 2	Engine fuel is not being filtered	Shut down engine or reduce power LAND AS SOON AS PRACTICAL		
CHIPS ENG 1, 2	Metallic chips detected in engine oil system	Shut down engine or reduce power LAND AS SOON AS PRACTICAL	Chips not present in oil	(2)
ENGINE	Used in production aircraft only			
FUEL TEMP HIGH	Fuel temperature is high	Switch boost pump ON. Note boost lights ON	Fuel temperature is below critical value	(3)
FUEL LOW FWD, AFT	Aft or forward fuel tank quantity is low. 30 min. of normal operation is possible when light first comes ON	Complete critical tasks quickly, then return to base	Tank contains fuel for more than 30 min operation	(4)
REFUEL VALVE OPEN	Pressure refueling control valve is OPEN - fuel cannot be transferred between tanks	Verify refuel valve control switch in pressure refueling control panel on right side of aircraft is in CLOSED position	Refueling valve is CLOSED	(5)
EXT TANKS EMPTY	Used in production aircraft only			
ECS	Environmental Control System (ECS) malfunction has been detected	Switch to standby ventilation and power-down non-critical electronics	ECS is operating normally	

NOTES:
(1) Light ON during start with cold oil is normal.
(2) Continued operation may result in severe engine damage.
(3) Hot fuel can not be drawn by engine-mounted suction pumps (vapor lock).
(4) Should both engines be crossfeeding from one tank, light ON indicates 15 min. of fuel remaining.
(5) Pressure refueling control panel access door will not close properly unless refuel valve control switch is in CLOSED position.

Table 8. (cont'd)

CAUTION LABEL	MEANING	CREW RESPONSE	LIGHT GOES OFF WHEN	NOTES
OIL LOW MN XMSN 1, 2	Main transmission is low on lube oil	One light - LAND AS SOON AS PRACTICAL Two lights - LAND AS SOON AS POSSIBLE	Quantity of oil is above minimum level	
OIL HOT MN XMSN 1, 2	Transmission lube oil cooling system malfunctioning	One light - LAND AS SOON AS PRACTICAL Two lights - LAND AS SOON AS POSSIBLE	Oil temperature does not exceed critical value	
CHIPS MN XMSN 1, 2	Metallic chips detected in main transmission oil system	LAND AS SOON AS POSSIBLE	Chips not present	(1)
OIL PRESS ACCESS PMP	Loss of oil pressure from accessory gearbox lube oil pump	If APU running and rotor system locked, shut down APU. If in flight. LAND AS SOON AS POSSIBLE	Oil pressure above unsafe value	(2)
PRI HYD	Failure or imminent failure of primary hydraulic system	LAND AS SOON AS POSSIBLE. IF OIL PRESS UTIL HYD caution light is also on then switch to EMERGENCY HYDRAULICS and LAND IMMEDIATELY	Primary hydraulic oil level and pressure is above unsafe value	(3)
OIL PRESS UTIL HYD	Failure of utility hydraulic system	LAND AS SOON AS POSSIBLE. If PRI HYD caution light is also on, then switch to EMERGENCY HYDRAULICS and LAND IMMEDIATELY	Pressure in the utility hydraulic system is above unsafe value	(4)
OIL LOW UTIL HYD	Utility hydraulic system is low on oil	LAND AS SOON AS PRACTICAL	Utility hydraulic system oil level is above unsafe value	(5)
TEMP HIGH TAIL GRBX	Excessive tail rotor gearbox temperature	LAND AS SOON AS POSSIBLE	Gearbox temperature is within normal limits	(6)
TEMP HIGH INTMD GRBX	Excessive intermediate gearbox temperature	LAND AS SOON AS POSSIBLE	Gearbox temperature is within normal limits	(6)
OIL PRESS NS GRBX 1, 2	Loss of pressure in indicated nose gearbox lube oil system	Shut down engine or reduce power. LAND AS SOON AS PRACTICAL	Nose gearbox oil temperature within safe limits	

NOTES:
(1) Continued operation could result in complete transmission failure.
(2) Accessory gearbox lube pump supplies oil to accessory gears and SDC for ground operation.
(3) Should the primary system fail and fluid quantity be low, all accessories, i.e., gun turret, flaps, etc, became inoperative.
(4) All accessories, i.e., gun turret, flaps, etc, became inoperative.
(5) Redundant flight control power remains available.
(6) Continued operation may result in tail rotor drive system failure.

Table 8. (cont'd)

CAUTION LABEL	MEANING	CREW RESPONSE	LIGHT GOES OFF WHEN	NOTES
OIL HOT NS GRBX 1, 2	Nose gearbox oil cooling system malfunctioning -OR- nose gearbox deicing equipment malfunctioning	Shut down engine or reduce power LAND AS SOON AS PRACTICAL Reduce power on affected engine. If applicable, switch NOSE GRBX anti-ice OFF	Nose gearbox oil temperature within safe limits	
CHIPS NS GRBX 1, 2	Metallic chips detected in indicated nose gearbox lube oil	Shut down engine or reduce power LAND AS SOON AS PRACTICAL	Chips not present	(1)
GEN 1, 2	Indicated AC generator is off the line. Opposite AC generator is carrying total AC load	Attempt resetting generator(s). One light - LAND AS SOON AS PRACTICAL. Two lights - LAND AS SOON AS POSSIBLE	AC generator is on the line and operating normally	(2)
RECT 1, 2	Indicated DC rectifier is off the line. Opposite DC rectifier is carrying total DC load	One light - LAND AS SOON AS PRACTICAL. Two lights - LAND AS SOON AS POSSIBLE	Rectifier is on line and functioning normally	(2)
TEMP HIGH RECT 1, 2	Excessive temperature detected in indicated DC rectifier	One light - LAND AS SOON AS PRACTICAL. Two lights - LAND AS SOON AS POSSIBLE	Rectifier temperature is within safe limits	
EXT PWR DR OPEN	Access door to external power receptacle is open - external power may be connected	Verify external power cable is disconnected and door is closed prior to moving aircraft	Access door is secure	
APU FAIL*	While APU is running: APU malfunction detected	Switch APU off. Attempt APU restart after waiting recommended time	APU is ready	
SDC	Shaft driven compressor output pressure is low	Be aware that dual engine restart is not possible without an operable SDC, unless ground pneumatic source is available	SDC output pressure is within tolerance	
APU ON	APU selector switch is in RUN or START position	APU should be shut down prior to take off	APU Selector Switch is in OFF position	
RTR BRAKE	Hydraulic pressure is being applied to the rotor brake caliper	LAND AS SOON AS POSSIBLE. Shut down both engines	Hydraulic pressure is not being applied to the rotor brake caliper	(3)

NOTES: (1) Continued operation may result in complete nose gearbox failure.
(2) Single failure will not impair aircraft operation. Dual failure results in automatic shift to battery power.
(3) Light ON is normal if the rotor brake control switch is in BRAKE or LOCK position.

*Pilot's caution panel only.

Table 8. (cont'd)

CAUTION LABEL	MEANING	CREW RESPONSE	LIGHT GOES OFF WHEN	NOTES
SAS	Stability Augmentation Subsystem (SAS) has become inoperative	If problems in control of aircraft are encountered, land as soon as practical	SAS is engaged and is operating normally	
BUCS FAIL	Back-up Control System (BUCS) failure has been detected	DO NOT engage BUCS		(1)
BUCS ON	BUCS mode has been selected by at least one crewmember	Opposite crewmember loses all control of aircraft		
FLAPS	Used in production aircraft only			
FORCE FEEL	Force feel function has become inoperative	Be aware that the longitudinal flight controls will not provide acceleration-related stick-feel		
ROCKETS	Used in production aircraft only			
CANOPY OPEN	Either canopy door is not positively CLOSED	Verify both canopy doors are CLOSED and SECURE	Both canopy doors are CLOSED	(2)
GUN DOWN	Used in production aircraft only			
IFF	Used in production aircraft only			
SS/TME	Used in production aircraft only			
VOICE CIPHER	Used in production aircraft only			
ENGINES CUT	Engine power cut to idle range by pressing ENG CUT button	Immediately initiate an autorotation to maintain N_R	Either crew member deactivates throttle chop feature by pressing ENG CUT button	(3)
DEICE HOT NS GRBOX	Overtemperature malfunction in nose gearbox anti-ice device	Switch OFF nose gearbox anti-ice device. Maneuver aircraft out of icing conditions, if possible	When temperature is below critical value	
OIL PRESS MN XMSN 1, 2	Loss of oil pressure in indicated main transmission lube system	One light - LAND AS SOON AS PRACTICAL Two lights - LAND AS SOON AS POSSIBLE	Oil pressure is above unsafe value	

NOTES: (1) Refer to WARNING on Page 1-62.
(2) Canopy doors must be securely closed to permit efficient crew and avionics cooling.
(3) THROTTLE CHOP ACTION IS RESERVED FOR TAIL ROTOR FAILURE EMERGENCY ONLY.

Table 9. Pilot and Copilot/Gunner Advisory Lights.

ADVISORY	LOCATION	COLOR	LIGHT COMES ON WHEN	LIGHT GOES OFF WHEN	NOTES
Fuel boost pressure engine 1, 2	Pilot's fuel control panel	Green	Fuel boost pressure is within tolerance	Fuel boost pressure is not present	(1)
Tailwheel status	Pilot's tail-wheel control panel	Green	Tailwheel is UNLOCKED	Tailwheel is LOCKED	
Aircraft standby power for engine overspeed system	Pilot's power quadrant	Green	Pilot performs specified test and system is operating normally	Pilot completes specified test	(2)
Ordnance safe	Pilot's and CP/G's armament panels	Green	Master arm switch is in SAFE position	Master arm switch is OFF or ARMED	
Ordnance armed	Pilot's and CP/G's armament panels	Amber	Master arm switch is in ARMED position	Master arm switch is OFF or SAFE	

NOTES: (1) Both lights should come ON during left engine start. One light should come on during right engine start.
(2) Labeled AC PWR.

PRESSURIZED AIR SYSTEM

56. The primary source of pressurized air is the shaft-driven compressor (SDC) driven by the main transmission accessory gear train. Engine bleed air is used as a primary pressurized and/or heated air supply for the engine air induction anti-icing system and soakback louver actuation. Bleed air from the left (No. 1) engine is the alternate source of pressurized air. In the event of a malfunction or failure in the SDC the system automatically switches over to the alternate source and the SDC caution light illuminates in both crew stations. Pressurized air is utilized for engine starters, fuel boost pump, ECU, engine cooling air louver actuators (in case of engine fire), and primary and utility hydraulic reservoirs. A receptacle for connecting a ground pneumatic source to the aircraft manifolding is provided. The receptacle is located on the left side of the fuselage, aft of the left engine. An adjacent receptacle enables ground personnel to power air-driven tools using aircraft-supplied pressurized air. All air-driven accessories may be operated from any of the pneumatic sources described. Failure of the SDC results in the impossibility of restarting both engines, unless a ground pneumatic source is available. However, a cross-bleed air start of the No. 2 engine is possible if the No. 1 engine is operating. The engine starter duty cycle is 2 minutes ON, 5 minutes OFF, or 3 automatic start cycles, and 30 minutes OFF. Figure 14 presents a schematic of the pressurized air system.

ENVIRONMENTAL CONTROL SYSTEM

Conditioned Air

57. The ECS, illustrated in figure 15, provides conditioned heating and cooling air to the crew stations and avionics bays. Air is supplied by an ECU driven by compressed air from the pressurized air system. The ECS is not to be operated on air supplied by an external source. The ECS is controlled with a selector switch marked ECS NORM, OFF, and STBY FAN, and a thermostat control. Both are located on the CABIN AIR panel in the pilot right console, illustrated in figure 4. During normal operation the selector switch is placed in the ECS NORM position and temperature is regulated by the thermostat control. Conditioned air is then ducted from the ECU forward to the pilot and copilot/gunner crew station. From there, air is drawn by a fan through a duct to the avionics bays, where it is used to provide avionics cooling, and then exhausted overboard.

58. ECS malfunctions or failures are indicated by the illumination of the ECS and/or SDC lights on the caution panels. An SDC malfunction requires no crewmember action, provided the No. 1 engine remains operating. However, if the ECS caution light comes on, the pilot must place the selector switch in the STBY FAN position. This action shuts off the ECU while permitting the fan to continue running. The copilot/gunner then opens auxiliary ventilation doors, one on each side of the forward crew station, thus allowing ambient air ventilation. These auxiliary doors must be closed for normal operation of the ECS. The ECS must be in the OFF or STBY FAN position during engine start.

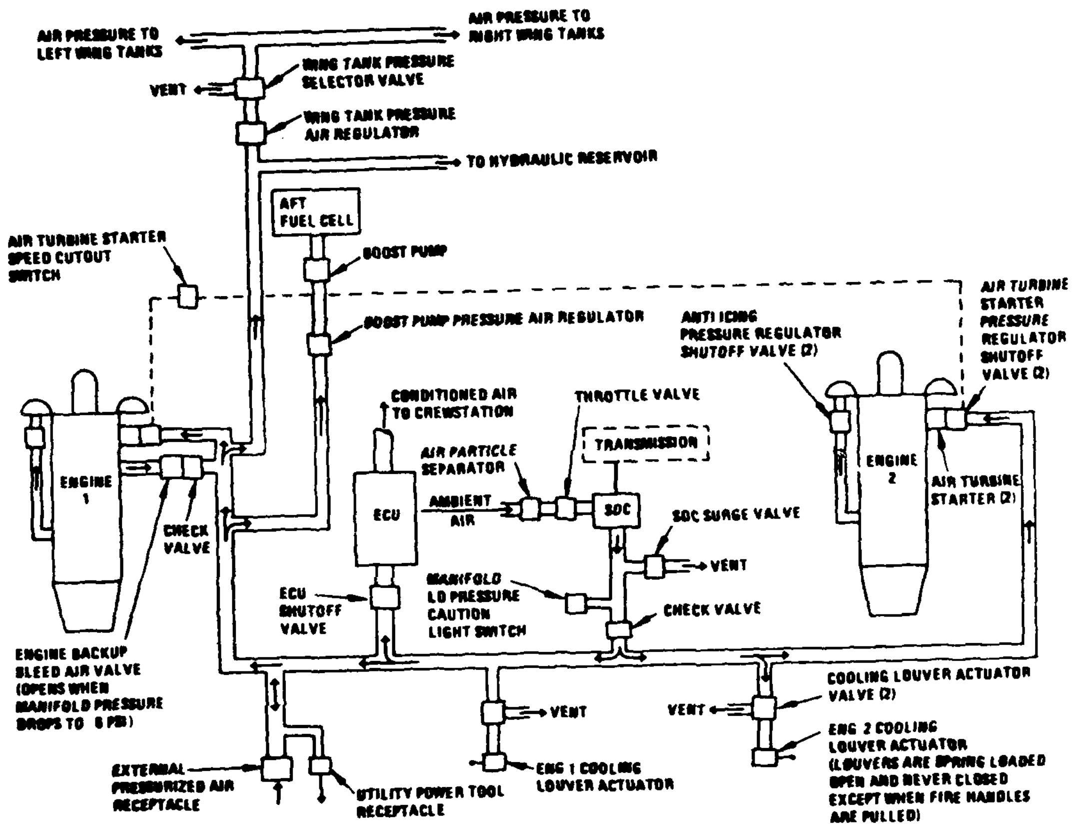


Figure 14. Pressurized Air Subsystem Schematic Diagram.

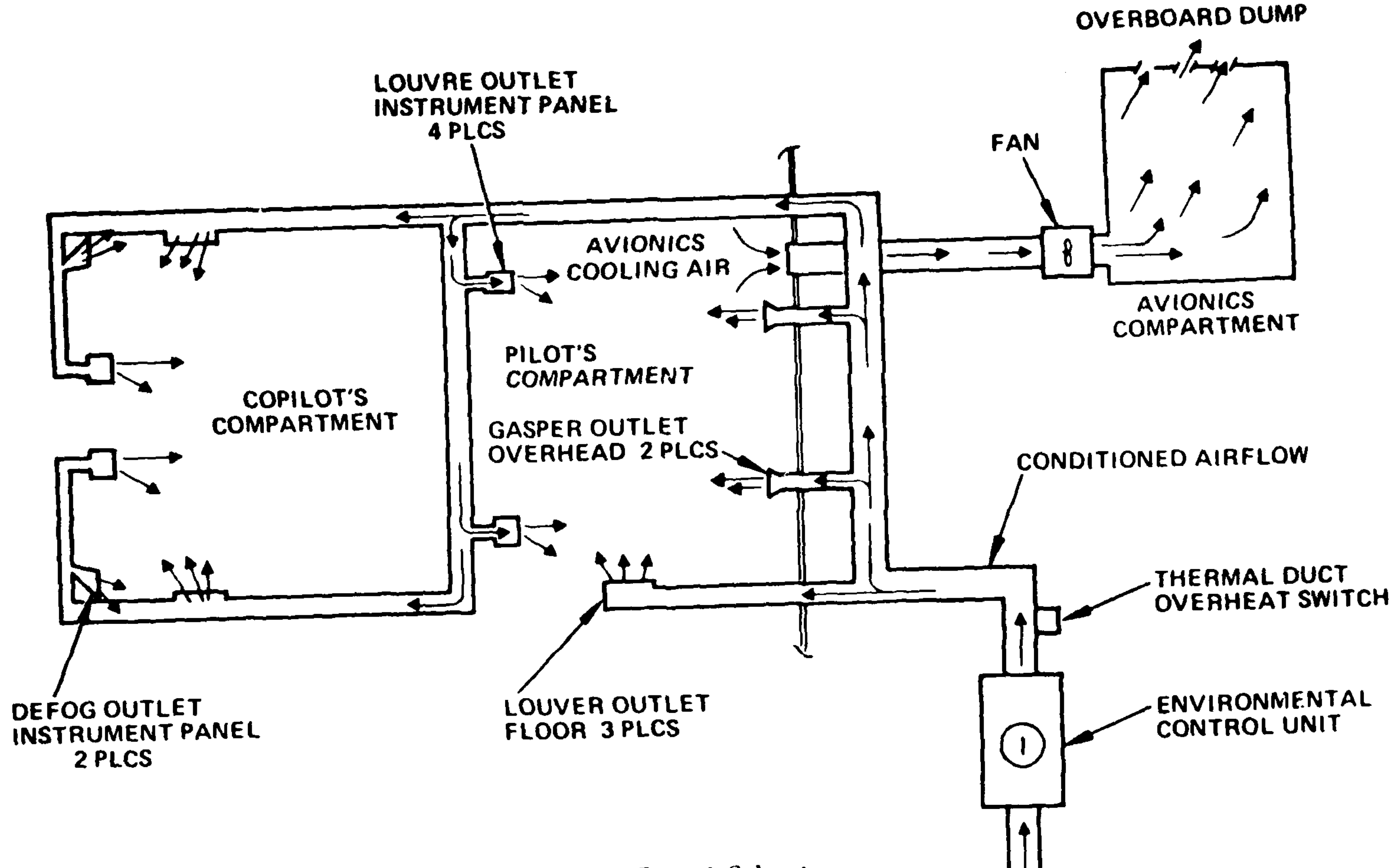


Figure 15. Environment Control Subsystem.

Rain Removal

59. Winshield wipers are provided for two windshields, one forward of and another directly over the copilot/gunner crew position. Both are electrically driven and controlled by a mode switch on the pilot anti-ice panel, located in his left console. The switch gives the pilot the ability to choose high or low wiping speeds, to turn the wipers off, and to drive them to a park position. The copilot/gunner is provided with a windshield wiper override switch on the anti-ice panel located in his left console. The switch, marked WSHLD WIPER, is normally in the PILOT position, allowing wiper operation to be controlled by the pilot windshield wiper control switch. When placed in the CP/G position the wipers are continuously driven at low speed, regardless of the position of the switch in the pilot crew station.

Anti-Icing

60. Anti-icing devices are provided for the pitot tube, engine inlets, and engine nose gearboxes. All anti-icing controls are located on the anti-ice control panels illustrated in figures 3 and 4. The engine air inlets are anti-iced with hot engine bleed air. Remaining anti-ice devices are of the electrical resistive heating type. All anti-ice devices must be switched on prior to operating in expected or known icing conditions.

Defogging

61. The canopy side panels are defogged with air delivered by the ECS. In the forward crew station, slits at the far left and far right edges of the copilot/gunner instrument panel direct a constant flow of air against the forward canopy side panels for defogging purposes. In the aft crew station, th pilot may defog his canopy side panels by adjusting the flow delivered by air registers in the left and right corners of his instrument panel or the air outlets located behind and above his crew seat.

EMERGENCY SYSTEMS

Fire Detection System

62. An optical surveillance fire detection system provides the crewmembers with instantaneous warning of the existence and location of fire in either engine comparment or the APU compartment. The system consists of sensors, amplifiers, and warning lights. Six sensors are provided, two in each compartment. The system uses three solid-state alarm amplifiers, one for each compartment. When an engine fire is detected, the appropriate instrument panel-mounted fire handle illuminates. APU fire causes the APU FIRE warning light to illuminate on the warning panel in both crew stations. Simultaneously, the APU fire handle in the right console of the aft crew station illuminates.

Fire-Extinguishing System

63. The fire-extinguishing system consists of bromotrifluoromethane (CF_3BR) stored at 600 psi in two containers. In the event of engine fire, either crewmember can activate the system by pulling the applicable FIRE PULL handle and placing the bottle selector switch in BTL 1 or BTL 2 position. Only the pilot has controls for extinguishing a fire in the APU compartment. When fire is no longer detected, the applicable handle and/or warning light goes out. A second discharge from the other bottle is available if needed.

Emergency Equipment

64. First-aid kits are provided at both crew stations and a portable fire extinguisher is located in the forward crew station.

Canopy Jettison System

65. A canopy jettison system is provided which allows emergency simultaneous jettison of all four canopy side panels, including the panels mounted in the crew station doors. The system utilizes shaped mild explosive charges with inboard shielding to protect the crewmembers from canopy spall. Three release handles are supplied. One is located in each of the crew station left upper instrument panels. The third is located behind a quick-release door above the nose of the aircraft just forward of the copilot/gunner windshield. All are painted bright yellow with black stripes. To activate the canopy removal system, any of the three handles is grabbed, squeezed to arm the system, and pulled to set off the charge which removes the canopy panels. Safety pins with red flags containing the words REMOVE BEFORE FLIGHT are installed when the aircraft is on the ground. All are stowed in the aircraft.

LIGHTING SYSTEM

Exterior Lighting

Landing Light:

66. A landing light is located in a fairing under the left side of the aircraft just forward of the landing gear strut attachment. The light is controlled by two toggle switches on the pilot and copilot/gunner collective stick grips, as illustrated in figures 3 and 4. One switch is used to extend or retract the light. The other has a rectangular thumb cap and is marked ON, OFF, and STOW. The light is fixed in azimuth but elevation can be adjusted with the extend-retract switch. The ON, OFF, and STOW switch must be in the ON or OFF position before the light can be extended or retracted. The light can be switched on while in the stowed position without damage from ,erheating.

Searchlight:

67. A searchlight is located in a fairing under the right side of the aircraft just forward of the landing gear strut attachment. The light is controlled by two toggle switches on the pilot and copilot/gunner collective stick grips. One switch is marked ON, OFF, and STOW. The other switch has five positions and is fitted with a round thumb cap. It is used to extend, retract, and control elevation and azimuth. The light can be articulated from the stowed position in elevation through 120 degrees and can be rotated through a ±180-degree azimuth. The ON, OFF, and STOW switch must be in the ON or OFF position to extend or retract the light. The light can be switched on while in the stowed position without damage from overheating.

Navigation, Formation, and Anticollision Lights:

68. Navigation lights are controlled by an ON/OFF switch located on the pilot exterior light control panel. The left and right lights are located in red and green transparent fairings on the wing tips and the aft white light is located on the end of the tail cone. Four formation lights, one on the upper surface of each wing, one on the upper center line of the aft fuselage, and one on the upper surface of the tail cone, are controlled by a rheostat on the pilot exterior light panel. A day/night high-intensity omnidirectional anticollision strobe light is located on each wing tip. A switch on the pilot exterior light panel, illustrated in figure 4, is provided to select white light for daytime operation or red for night.

Interior Lighting

69. Interior lighting consists of instrument lighting, panel lighting, utility lights, inspection and maintenance lights, and emergency floodlights. Interior lighting control panels for both the pilot and copilot/gunner are illustrated in figures 3 and 4. The red primary instrument lighting system is integral with the instruments. Intensity is controlled in two lighting zones by controls on the interior lighting panel at each crew station. Moving the flight instrument (FLT INST) dimming control in either crew station out of the OFF position switches that crew station's caution, warning, and advisory lights to the dim mode. A secondary selectable red/white emergency floodlighting system is installed under the glare shields of the instrument panel. The floodlight selector switches are also located on each interior lighting panel. The avionics and aircraft systems control panels are integrally illuminated. Two dimming controls on the interior lighting panels are provided. Two service panels provide electrical power for maintenance and inspection lights. They are located on the right side of the aircraft. One is adjacent to the forward crew station and the other is aft of the battery compartment.

AVIONICS SYSTEM

70. Avionics equipment installed in the aircraft includes two intercom control systems, C-6533/ARC; one radio set, ARC-115 VHF/AM; one radio set, ARC-116 UHF/AM; an ARTSS; and a VHF/UHF antenna mounted on the underside of the aircraft.

Communications Control

71. An intercom set (ICS) is provided in both crew stations. The set contains switches and controls for the following functions: receive UHF, transmit UHF, volume control; receive VHF, transmit VHF, hot mike. Intercom channel 2 connects to the UHF radio, while channel 3 connects to the VHF radio. Channels 1, 4, and 5 are reserved for future growth. Placards are provided showing the available functions of the ICS. The controls for the AN/ARC-115 VHF/AM radio and AN/ARC-116 UHF/AM radios are located on the pilot right console. Two external connectors are provided, enabling maintenance personnel to communicate with the pilot and copilot/gunner during ground operations. Three-position rocker-type microphone keying switches are located on each cyclic stick grip (fig. 16). The switches are spring-loaded off and have positions marked ICS and RADIO. Additional foot-operated switches are provided to transmit in the mode selected on the ICS.

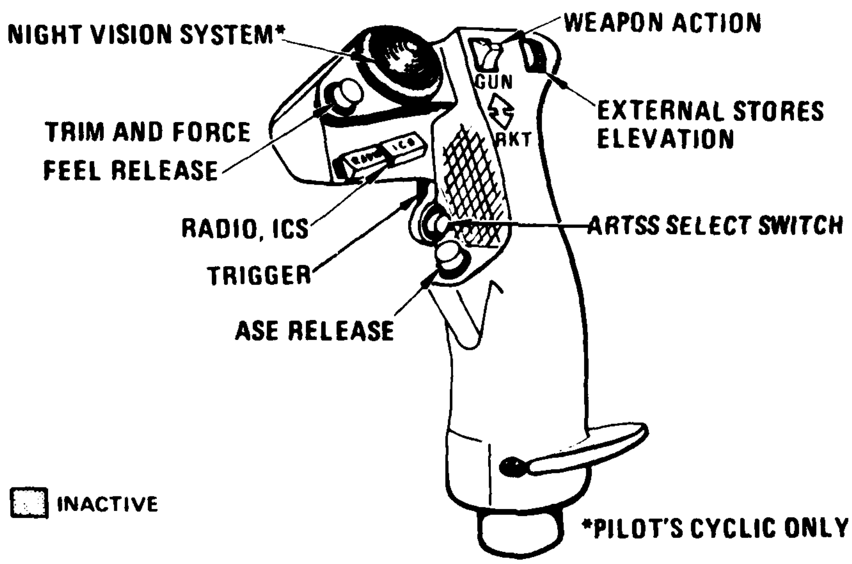


Figure 16. Cyclic Stick Grip.

Automatic Radio Transmitter Select System

72. The ARTSS allows the pilot to automatically switch between three radio transmitters by pressing the ARTSS button on his cyclic stick grip, illustrated in figure 16. A display is provided above his intercom control panel to indicate the transmitter selected. To use the ARTSS the pilot selects channel 5 on the intercom control panel and sequences transmitters via the cyclic-mounted ARTSS pushbutton. Manual transmitter selection may be utilized at any time.

ARMAMENT SYSTEM FOR FLIGHT TEST AIRCRAFT

Area Weapon

73. The YAH-64 area weapon is a single-barrel, automatic, externally powered chain-driven 30mm gun. The system utilizes a flexible turret with ±110 degree azimuth and +13.2-degree, -60 degree elevation coverage. Provisions have been made for a linkless ammunition storage and feed system with a capacity during the current program phase of greater than 90 rounds. Capacity of the production aircraft is 1200 rounds. The gun's rate of fire is selected for optimum airframe compatibility and is currently 535 rounds (±20) per minute. Figure 17 illustrates area weapon system components.

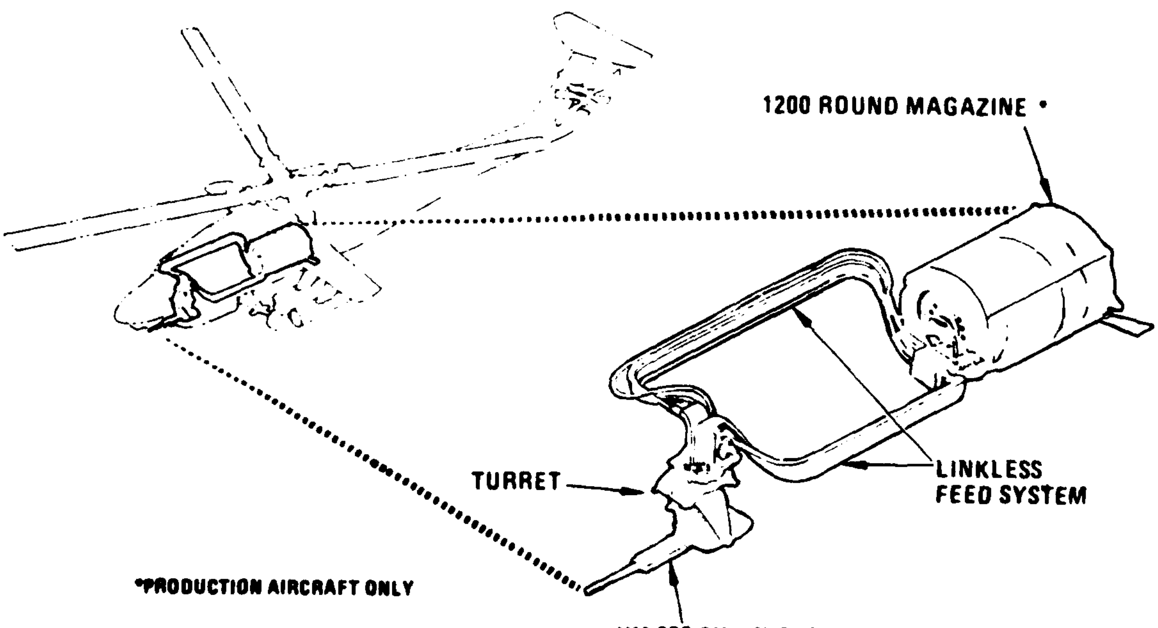


Figure 17. Area Weapon Subsystem.

Aerial Rocket Subsystem

74. The 2.75-inch FFAR are managed and fired by an on-board rocket control system. As many as four rocket launchers may be carried on any one flight. Each accommodates up to 19 FFAR. Figure 18 presents the location of aerial rocket system components.

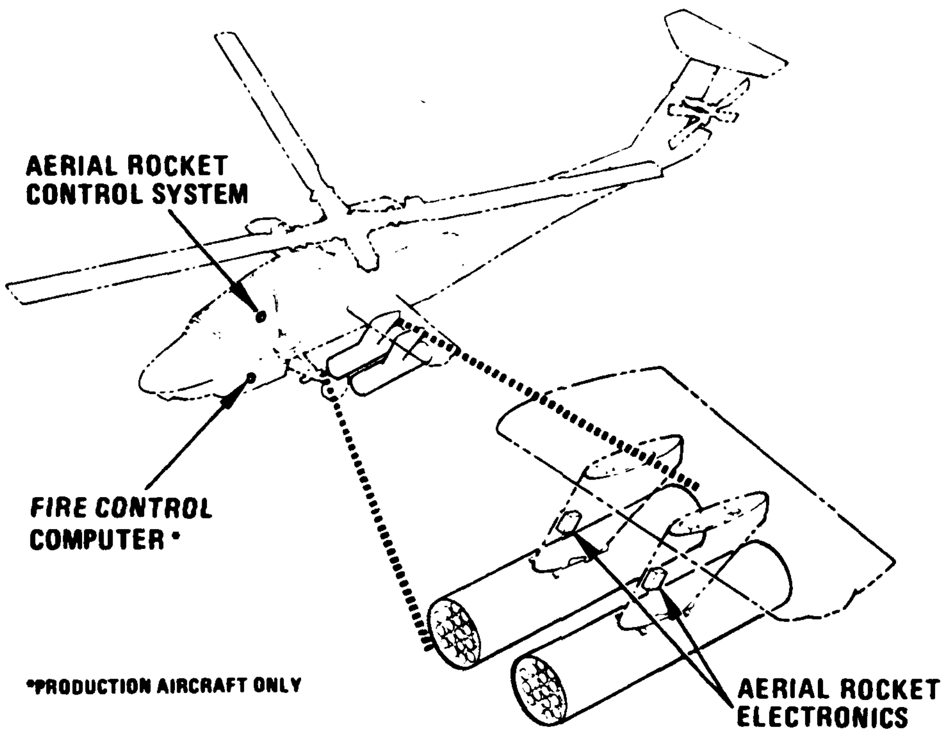


Figure 18. Aerial Rocket Subsystem.

Point Target Missile Subsystem

75. The YAH-64 is intended as an aerial launch platform for a point target missile system. The production aircraft will include a complete fire control system operable from the copilot/gunner crew station tracker, night target acquisition system, and laser rangefinder. Figure 19 illustrates anticipated point target missile equipment.

Stores Jettison

76. An emergency wing stores force jettison system is provided. The pilot may selectively jettison wing stores. The copilot/gunner may select a complete jettison of all wing stores. Stores jettison control panels for the pilot and copilot/gunner are illustrated in figures 3 and 4.

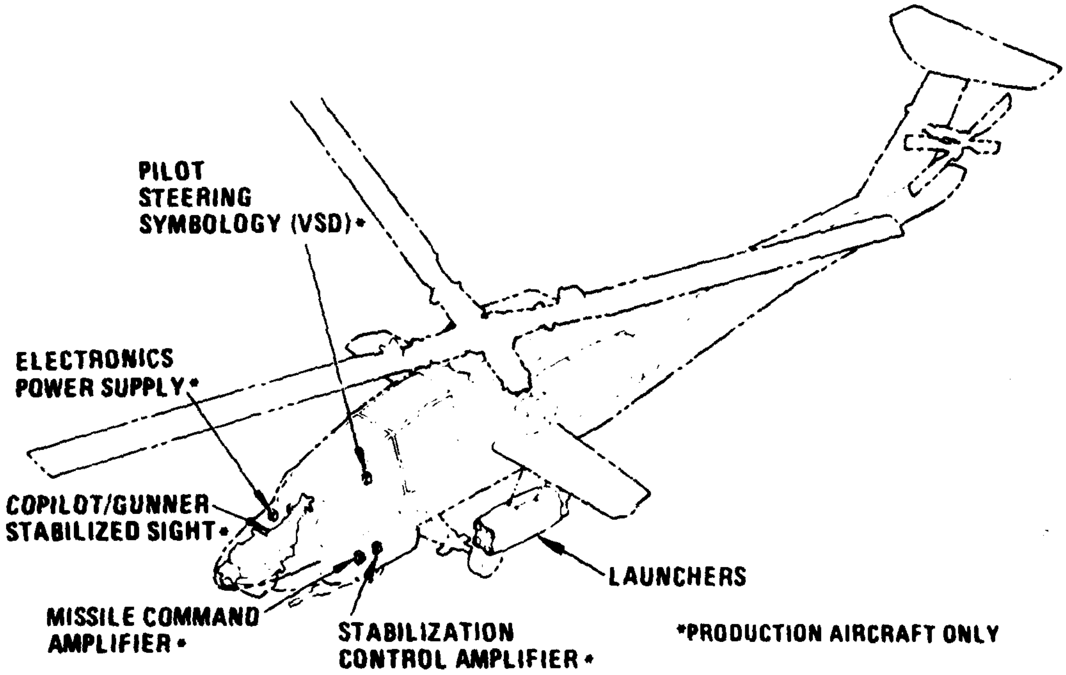


Figure 19. Anticipated Point Target Missile Subsystem.

Armament Controls

77. The pilot has authority over weapons system utilization. A MASTER ARM switch on his armament panel, illustrated in figure 4, controls weapon system modes. In the OFF position, all systems are switched off. In the SAFE position, those systems selected by the pilot or the copilot/gunner may be energized by subsequent switch action up to but not including their triggers. In the ARM position, all weapons which have been selected by either crewmember are "hot" and may be fired. A green advisory light comes on when the MASTER ARM switch is in the SAFE position. A yellow light comes on when the switch is in the ARMED position. Circuit breakers provide a redundant means of safe-ing all ordnance.

78. A three-position switch marked GUN is placed in the fixed forward (FIX FWD) position to enable the pilot to fire the 30mm gun at zero-degree azimuth and zero-degree elevation. The position marked HMS, for production aircraft purposes, is used to permit the crew to fire the gun at azimuth and elevation angles chosen by the copilot/gunner.

79. A second three-position switch marked RKT (rocket) is placed in either the HOVER or the FWD FLT (forward flight) position to permit the crew to launch 2.75-inch FFAR. The HOVER position is used for firing the rockets only at airspeeds less than 40 KIAS. With the MASTER ARM switch in either SAFE or ARM, the three-position switch marked ROCKET positions the pylon stores as follows:

a. With the PILOT or CP/G switch in HOVER position, the crew can move the pylon stores up or down at any airspeed.

b. With the PILOT or CP/G switch in FWD FLT position, the pylon stores are automatically positioned and locked in the flight stow position.

80. With the MASTER ARM or both pilot and copilot/gunner ROCKET switches in the OFF position, the pylon stores are automatically positioned and locked in the ground stow position at airspeeds below 40 knots and the flight stow position at airspeeds above 40 knots. The pushbutton marked WIRE CUT and the two rotary switches marked HMS RETICLE and RKT RANGE are inactive.

81. Three major armament system controls are located on the cyclic stick grip illustrated in figure 16. A three-position momentary action thumb-operated toggle switch located on the right upper portion of the grip permits the crew to change external stores elevation (POD ELEV). The switch is pushed by the thumb to lower the elevation angle and pulled to raise the elevation. The POD ELEV switch is inoperative unless the RKT switch on either armament panel is in the HOVER position and the MASTER ARM switch is not OFF. Figure 20 illustrates external stores positioning logic.

82. A weapon action (WPN ACT) switch is located to the left of the POD ELEV switch. This three-position momentary action toggle is provided with a distinctive thumb cap to permit identification by touch. When pushed upward by the thumb,

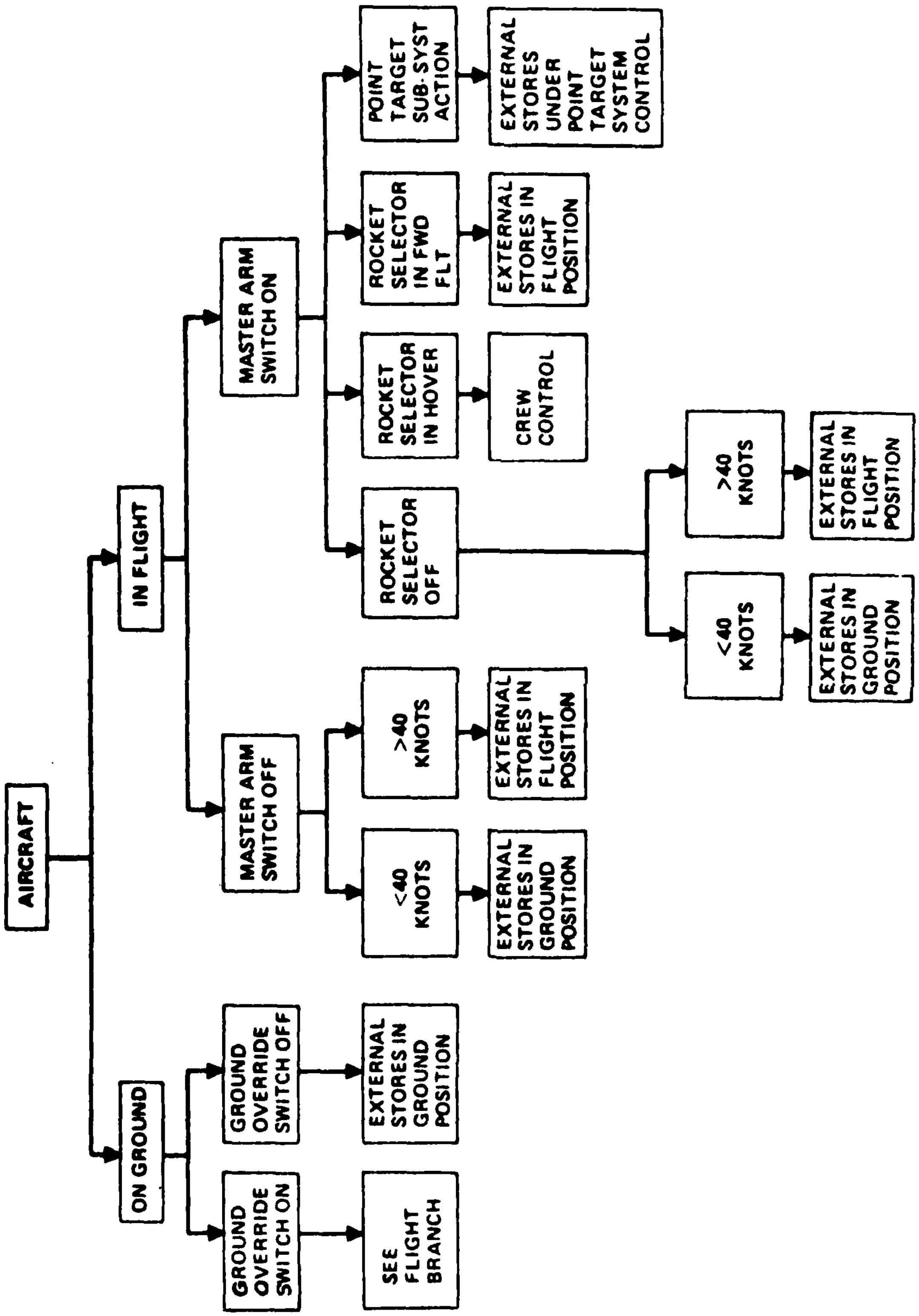


Figure 20. External Stores Position Logic.

gun turret pointing circuits are energized, provided the GUN switch on the armament panel is selected and the MASTER ARM switch is not in the OFF position. When pulled down, rocket action circuits are energized, provided the RKT and the MASTER ARM switches on the armament panel are not in the OFF position. During rocket firings with the rocket selector in the HOVER position and the rocket launcher depressed $\geq$ 10 degrees, the wing flaps are commanded 20 degrees up when weapon action is taken. The weapon action switch should always be operated prior to operating the trigger to prevent flap damage.

83. A guarded trigger switch is located on the forward side of the cyclic grip. To activate the switch, the guard is pushed forward against a spring by the right index finger. The trigger is then squeezed and pulled rearward through a detent to activate the trigger circuit. Under normal operating conditions, before any armament system can be activated to fire a round or launch a missile or rocket, the sequence shown below must not have been broken. Failure to set up the system properly prior to trigger activation will break the chain of built-in safeties and prohibit the weapon from firing.

a. Aircraft must be airborne.

b. MASTER ARM switch must be in ARMED position.

c. Appropriate weapon selector on armament panel must be in correct position.

d. WPN ACT switch must be held in desired position.

e. Trigger must be squeezed while holding WPN ACT switch in desired position.

84. The copilot/gunner is provided with an EMERGENCY ARM switch located on his armament panel, which is illustrated in figure 3. It is used primarily in emergencies. Two advisory lights, one green and one yellow, advise him of the safe or armed status of the aircraft armament systems as selected by the pilot, or of the status of his own armament systems in an emergency.

85. A guarded switch marked PLT/GRD OVRD (pilot/ground override) gives the copilot/gunner the ability to override the pilot MASTER ARM switch and safe all ordnance in an emergency. It also allows armament systems to be fired while the aircraft is on the ground by overriding a ground weight-on-the-wheels safety switch. With the PLT/GRD OVRD switch in the ON position, only the copilot/gunner EMERGENCY ARM switch selects armament system mode for all aircraft ordnance.

86. The copilot/gunner armament control panel contains a gun-rounds-remaining (GUN RDS REMAIN) device which functions as a burst length programming switch. In normal use the number of rounds loaded is set on the indicator and it counts down to zero as the area weapon is fired. In the flight test configuration

the copilot/gunner could set any desired number of rounds on the counter (up to the quantity actually remaining). When the counter reached zero, firing automatically terminated. This capability was included to facilitate testing. The actual ammunition status was calculated by the crew from the takeoff load and amount fired. The MV CAL pushbutton, HMS RETICLE, and RKT RANGE rotary switches and the TOW switch were inactive.

87. A weapons control (WPN CONT) panel, illustrated in figure 3, is provided on the copilot/gunner instrument panel, to the right of his armament panel. Controls thereon allow him to adjust 30mm gun azimuth and elevation angles. During flight testing it replaced point target system specialized controls which occupy the same relative space in the production aircraft. Position control of the area weapon turret is exercised by setting the desired azimuth and elevation positions on the appropriate dials, placing the MASTER ARM switch in SAFE or ARM position, selecting HMS on the pilot armament control panel (or HMS/SS on CP/G), and operating the cyclic-mounted action switch. At this time the weapon will point to the position selected. Upon release of the action switch (or turning off either of the switches listed above) the turret will return to stow position only in elevation (azimuth position will not change). The position of the turret may be changed by changing the dial settings while the action switch is depressed, if desired. For flight testing the turret was not lowered less than 40 degrees with azimuth angles greater than 80 degrees. As a reminder to crewmembers the TUR AZ control knob was striped yellow from 80 to 110 degrees left and right. The TUR EL control knob was striped yellow from 40 to 60 degrees down.

APPENDIX C. FLIGHT CONTROL DESCRIPTION

GENERAL

1. The YAH-64 helicopter employs a single hydromechanical irreversible flight control system with an electrohydraulic backup system. The hydromechanical system is mechanically activated with conventional pilot cyclic, collective, and pedal controls through a series of push-pull tubes going to four airframe-mounted hydraulic servo actuators. The four hydraulic servo actuators control longitudinal cyclic, lateral cyclic, collective, and tail rotor collective pitch, and are powered by two independent 3000-psi hydraulic systems which are powered by hydraulic pumps mounted on the accessory gearbox to allow full operation under a dual-engine failure condition. An ASE system is installed to provide closed-loop rate stability augmentation control (SAS) of limited 10 percent authority. Included with the ASE are a FAS, a wing flap control system, and a fly-by-wire emergency backup control system (BUCS) which was not connected during this evaluation. An FTS is incorporated in cyclic and pedal controls to provide a control force gradient with control displacement from a selected trim position. A force trim interrupter button, located on the cyclic grip, provides a momentary interruption of the force trim in all axes simultaneously to allow the cyclic or pedal control to be placed in a new trim position. A lateral-cyclic 1g spring is incorporated to reduce cyclic control imbalance characteristcs with the FTS OFF. The collective lever has a mechanical friction device and a 1g balance spring to balance the collective control forces. Full control travel is 9.5 inches in the cyclic longitudinal control, 9 inches in the lateral control, 11.8 inches in the collective control, and 6.5 inches in the rudder pedals.

CONTROL SYSTEMS

Cyclic Control System

2. The cyclic control system consists of dual-tandem cyclic control sticks attached to individual support assemblies at the cyclic stick base (fig. 1). The support assembly houses the primary longitudinal and lateral control stops, a lateral 1g spring to reduce stick imbalance characteristics, and two linear variable displacement transducers (LVDT) designed to measure electrically the longitudinal and lateral motions of the cyclic for ASE computer inputs. A series of push-pull tubes and bell cranks (fig. 2) transmits the motion of the cyclic stick to servo actuators and the mixer assembly. Motion of the mixer assembly (fig. 3) positions the nonrotating swashplate, which is linked to the rotating swashplate to control the main rotor blades in cyclic and collective pitch. The cyclic stick uses the standard Army handgrip (fig. 29, app B).

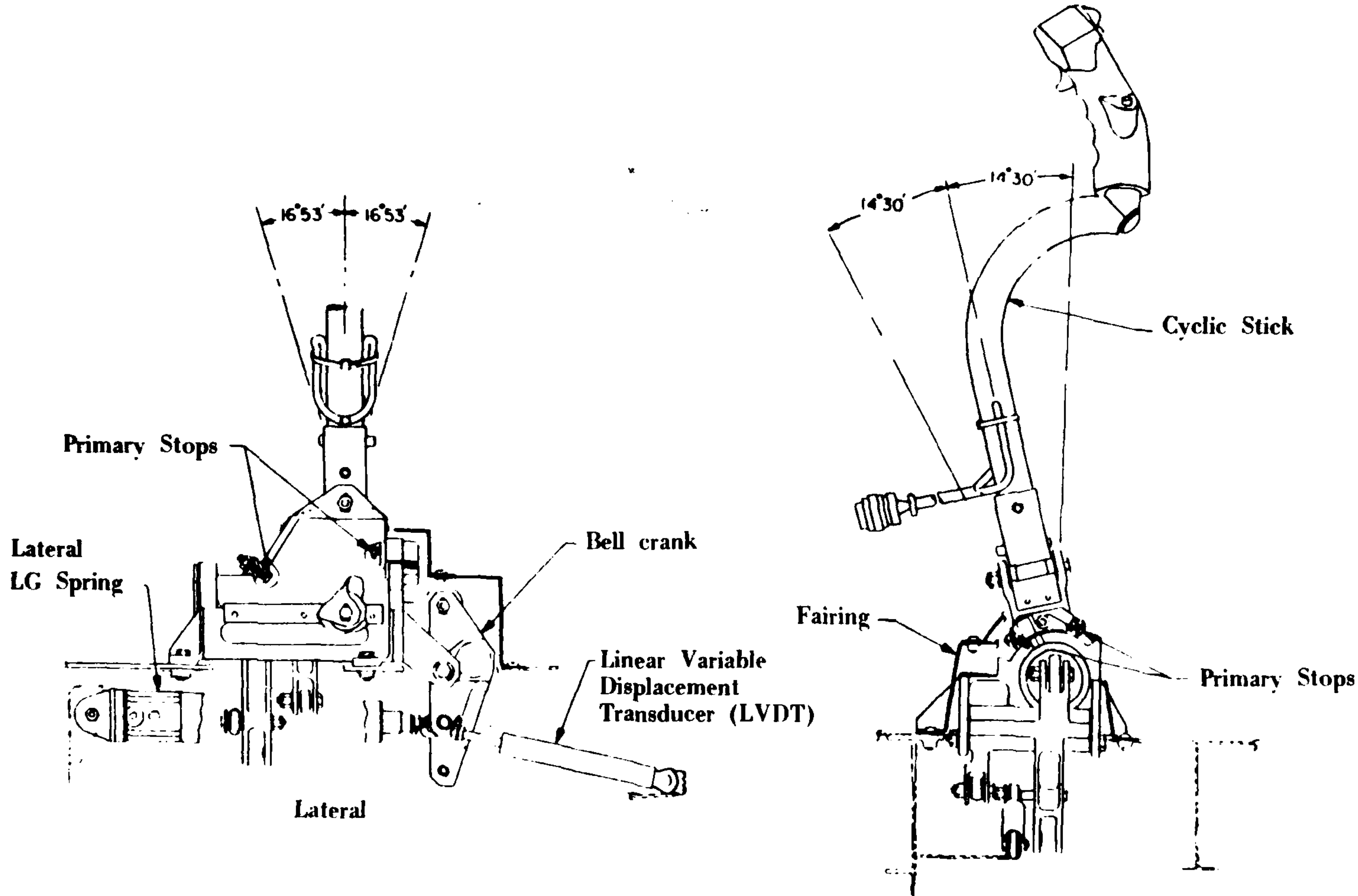


Figure 1. Cyclic Stick Schematic.

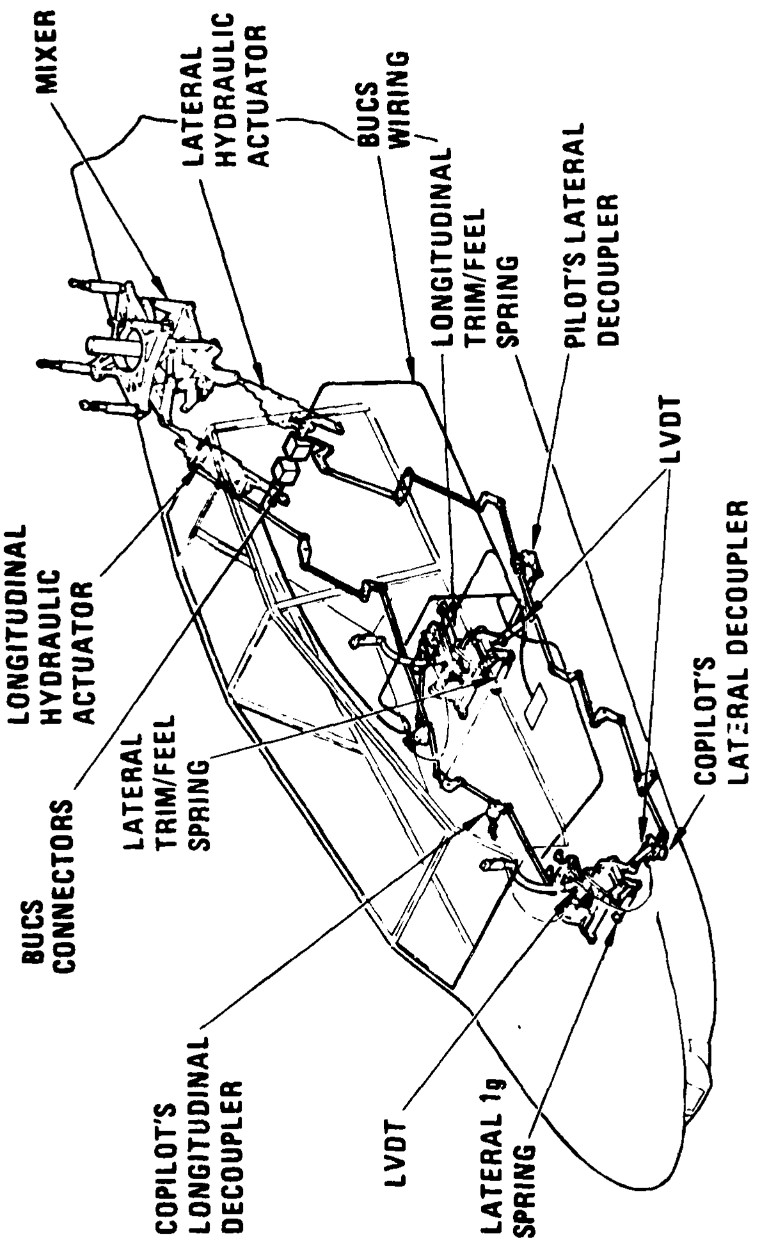
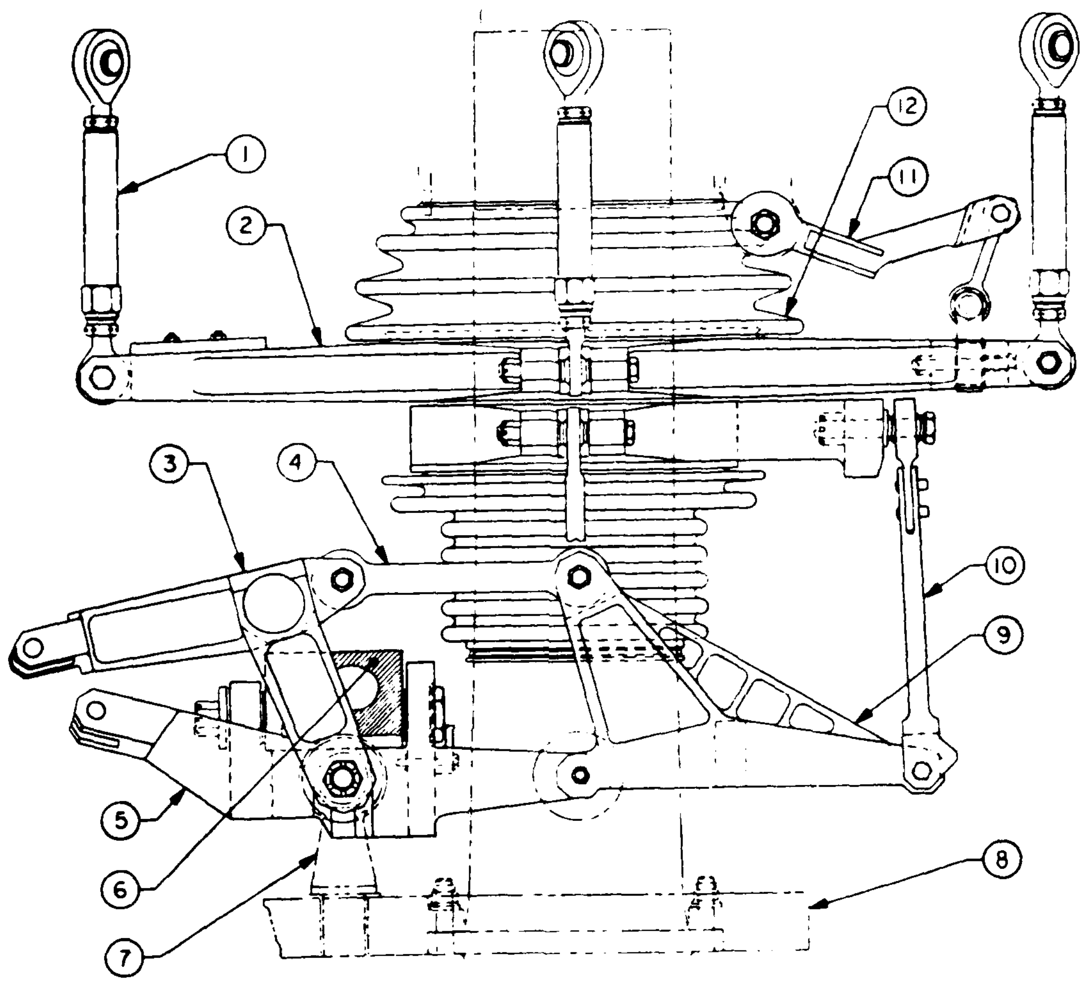


Figure 2. Longitudinal and Lateral Control System.



1. PITCH LINK ASSEMBLY
2. SWASK PLATE ASSEMBLY
3. LONGITUDINAL INPUT BELLCRANK
4. LONGITUDINAL LINK
5. COLLECTIVE INPUT BELLCRANK
6. LATERAL BELLCRANK
7. MIXER SUPPORT
8. MAST SUPPORT STRUCTURE
9. LONGITUDINAL DIFFERENTIAL INPUT BELLCRANK
10. TORQUE LINK
11. ROTATING SCISSORS ASSEMBLY
12. ROOT

Figure 3. Swashplate Mixing Assembly.

Force Trim System

3. The cyclic control FTS provides cyclic control stick feel and allows close repositioning with the use of the cyclic trim button (fig. 29, app B). Individual longitudinal and lateral electromagnetic brake clutches incorporating trim feel springs are provided for stick centering and a stick force gradient. Additionally, a toggle switch is provided on the pilot left console to activate or disable the trim system. The electromagnetic brake clutch is powered by 28 VDC and is protected by a trim circuit breaker panel. In the event of complete DC failure, both the FTS and FFS are disabled, and cyclic stick movement will not be resisted fore and aft but will be held laterally by the 1g spring.

Collective Control System

4. The collective pitch control system consists of dual-tandem control sticks connected by push-pull tubes and bell cranks (fig. 4). Located at each collective stick base assembly unit are the primary control stops, a 1g balance spring to counteract the collective control forces of the stick and linkages, and an LVDT (fig. 5). The LVDT supplies electrical inputs to the ASE and to the load-demands spindle of the engine hydromechanical unit (HMU), which is a section of the fuel control unit for the YT700-GE-700 engines. The input to the HMU provides collective pitch compensation which acts as a main rotor droop compensator. Additionally, the collective LVDT provides inputs to the wing flap control system. A series of push-pull tubes and bell cranks transmits collective movements to the collective servo actuator, the main rotor mixer unit, and the main rotor.

5. A switch box assembly at the top of each collective pitch stick contains numerous switches, as depicted in figure 2a, appendix B. Among them is one nonstandard switch. The engine-cut button provides for rapid deceleration of the engines to ground-idle in event of an emergency where the pilot cannot release the collective control to retard the speed selectors. Both collective sticks incorporate adjustable friction devices.

Directional Control System

6. The directional control system (fig. 6) consists of the following components: two sets of adjustable directional control pedals with a total adjustment of 6 inches, accomplished through the use of pedal adjust knobs located between the pilot/copilot's feet (fig. 7); two sets of wheel brake cyclinders; and a series of push-pull tubes and bell cranks which extend the length of the airframe via the tail rotor servo actuator and terminate at the tail rotor gearbox. Attached to each directional pedal assembly are the primary tail rotor control stops and one LVDT.

Pedal Trim System

7. The pedal trim gradient system incorporates the same magnetic clutch and spring assembly as previously described for the cyclic stick (para 3). The trim gradient is high to help reduce control sensitivity.

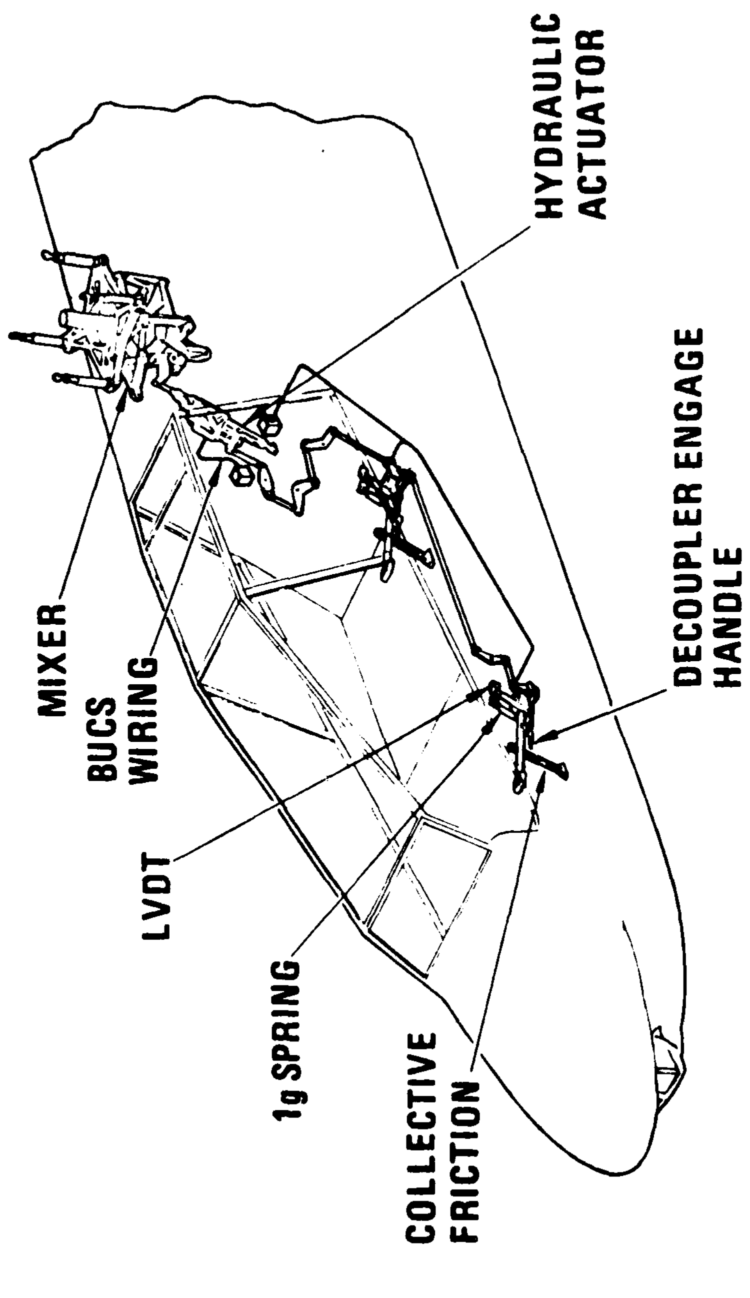


Figure 4. Collective Control System.

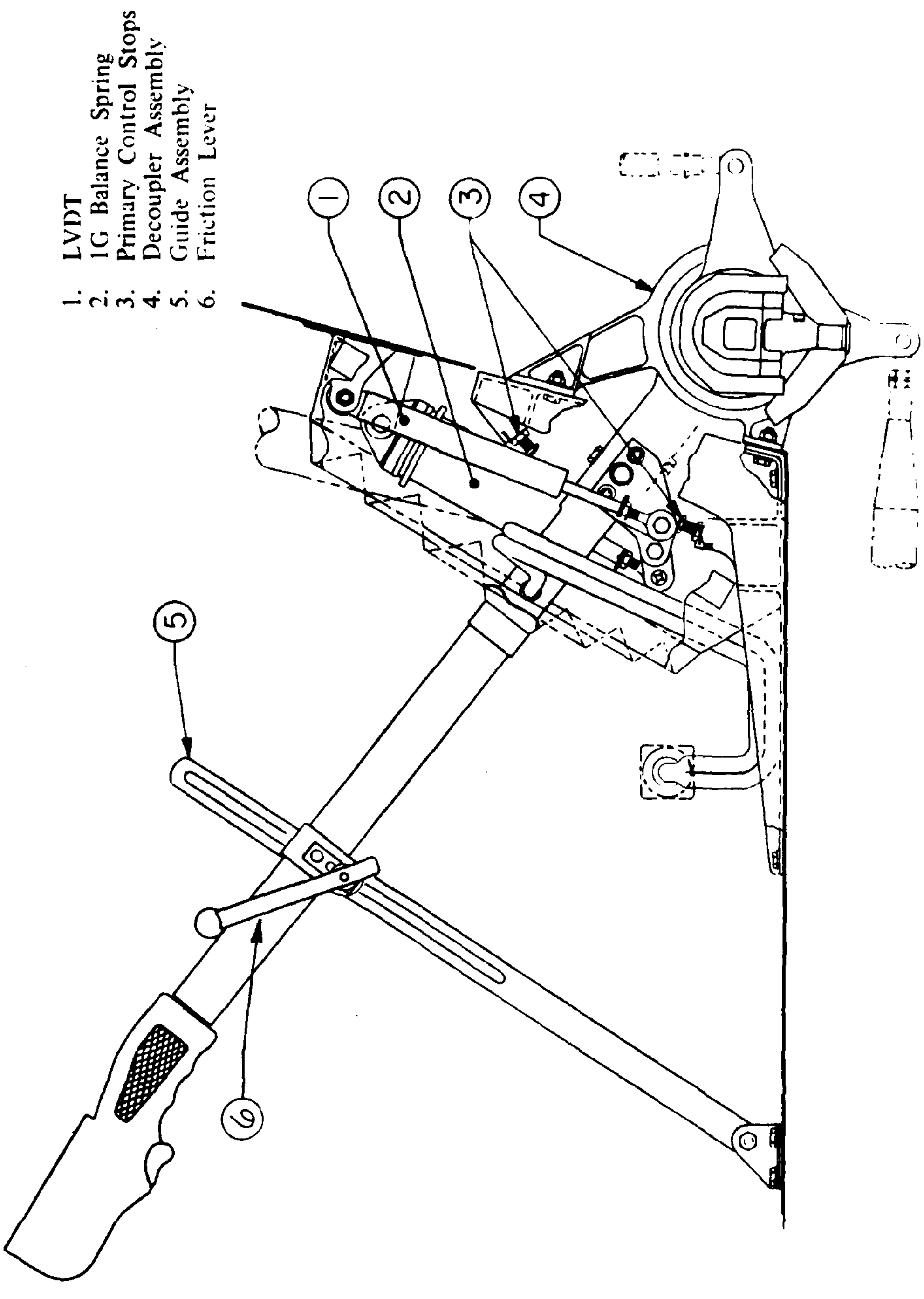


Figure 5. Collective Stick Schematic.

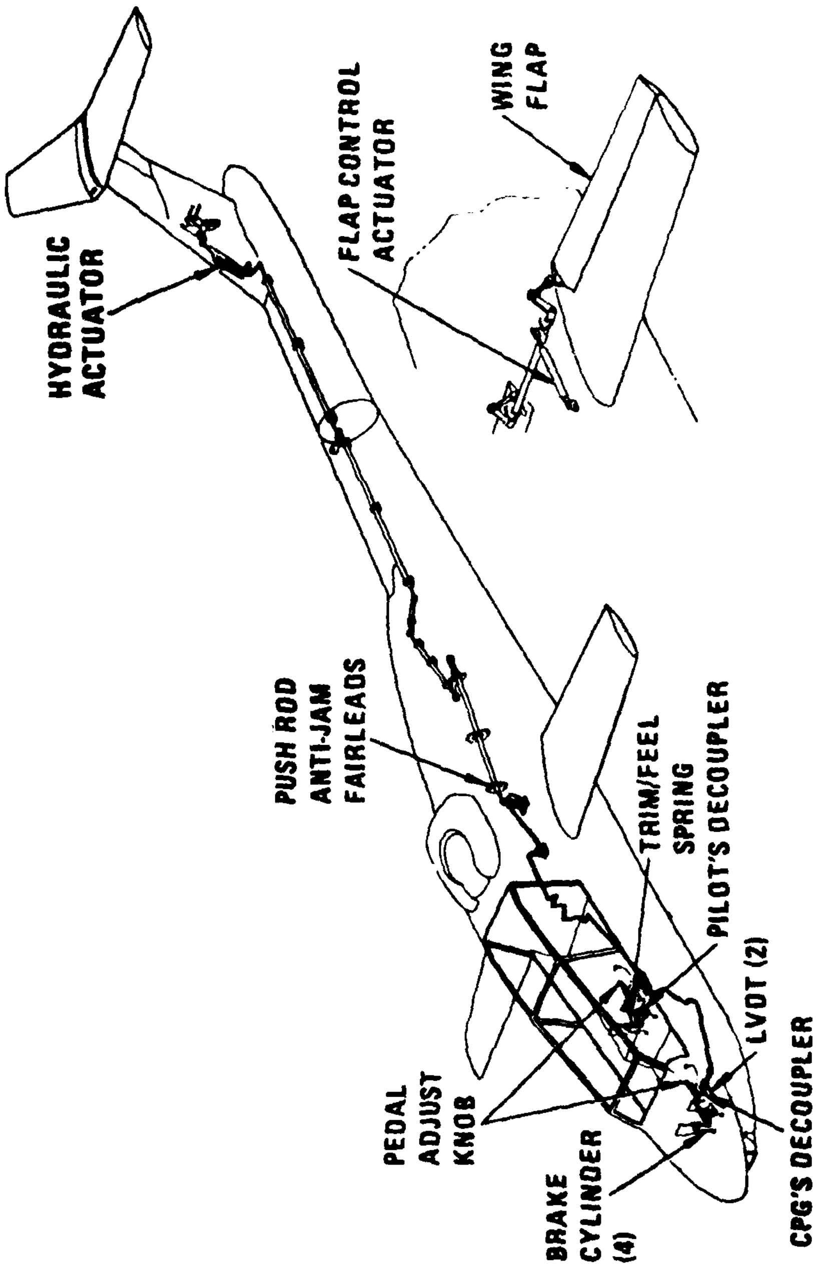


Figure 6. Directional Control System.

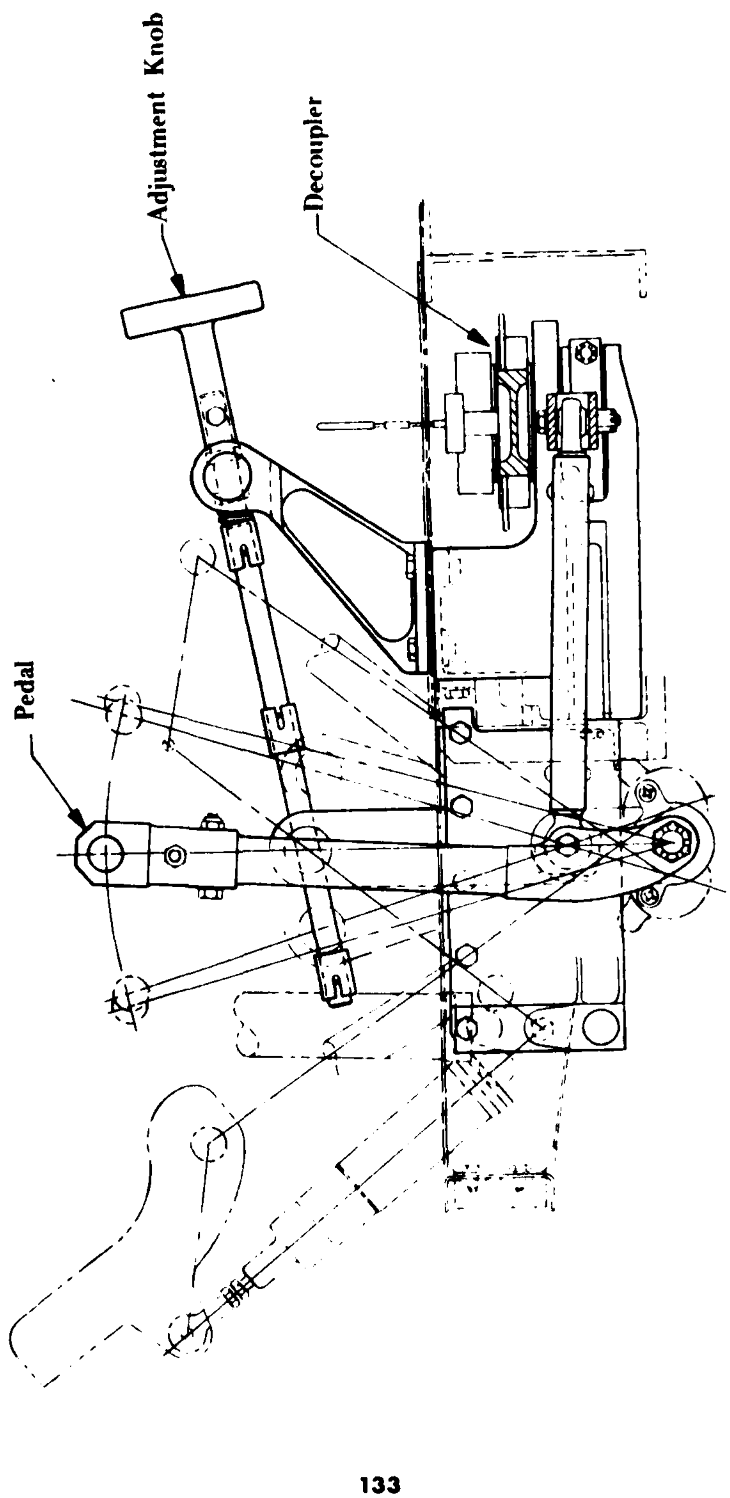


Figure 7. Directional Pedal Assembly.

HYDRAULIC SYSTEM

General

8. The hydraulic system consists of four hydraulic servo actuators powered simultaneously by two independent 3000-psi hydraulic systems. The two systems (primary and utility) are driven off the accessory gearbox utilizing variable displacement pumps, independent reservoirs, and accumulators (fig. 15, app B). The APU drives all accessories, including the hydraulic pumps, when the aircraft is on the ground and the rotor is not turning.

Primary Hydraulic System

9. The primary hydraulic system consists of a 0.52 quart-capacity reservoir, which is air charged to 20 to 60 psig using air from the shaft-driven compressor; an accumulator, which has a nitrogen precharge of 1600 psi, designed to reduce surges in the hydraulic system; and a primary manifold that redirects the fluid to the lower side of the dual-tandem actuators of the four servo actuators and SAS (fig. 15, app B). The primary system provides the hydraulic pressure for the SAS.

Utility Hydraulic System

10. The utility hydraulic system consists of a 1.16-gallon reservoir and a 3000-psi accumulator to drive the APU starting motor and to provide hydraulic pressure to the flight control system in the event of a dual hydraulic system failure. Each servo actuator simultaneously receives pressure from the primary and utility systems to drive the dual-tandem actuators. This design allows the remaining system to automatically continue powering the servos in the event of failure of a hydraulic system. The utility manifold directs fluid to the upper side of the servo actuators, the stores pylon system (which actuates the pylons in elevations from +10 to -28 degrees), wing flaps, gun azimuth and elevation, and rotor brake. Other manifold functions include a low fluid sensor which isolates all auxiliary functions to provide hydraulic pressure only to the servo actuators and rotor brake and a low-pressure sensor which isolates the accumulator to remain as a reserve hydraulic source for the servo actuators.

AUTOMATIC STABILIZATION EQUIPMENT SYSTEM

11. The ASE system (fig. 8) consists of four components: an ASE computer controlling the SAS, a control programming unit for the wing flaps, the FFS, and the BUCS. The BUCS was not operational during this evaluation.

12. The ASE computer receives flight control inputs via the individual LVDT's; the pitch, roll, and yaw rate gyros; a vertical gyro; a lateral accelerometer; and an airspeed sensor. The analog computer integrates all inputs to provide smooth control signals for the SAS control servos. Additionally, the computer has a built-in

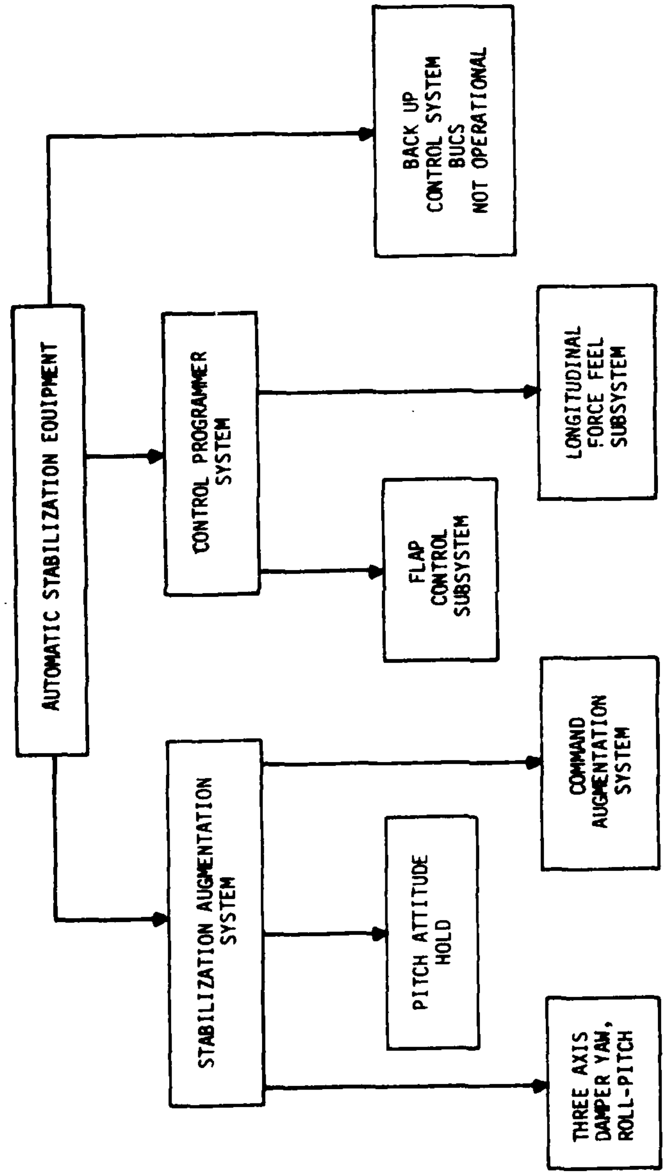


Figure 8. Automatic Stabilization Block Schematic.

test equipment (BITE) system. The BITE allows the pilot to check the automatic hardover monitoring circuits prior to rotor engagement. Pilot-to-ground BITE check procedure is to quickly apply small control inputs to individually decouple the pitch, roll, or yaw SAS channels. Disengagement of each channel indicates the ASE hardover protection system is operable. Additionally, in flight the automatic servo monitor system is designed to compare stick rate to stick displacement, disabling a particular SAS channel if the two are not compatible (hardover protection) (fig. 9). The FFS ground check is to verify that there is a fore and aft stick force with cyclic displacement. ASE cockpit control switches are provided to the pilot only; however, SAS disengagement switches are located on both cyclic stick grips (fig. 29, app B).

Stability Augmentation System

13. The SAS is a three-axis limited authority (10 percent) angular rate referenced damper. Aircraft stability is achieved through the use of three rate gyros providing closed-loop inputs to the servo actuators. A vertical gyro provides inputs designed to improve the longitudinal static stability by employing a weak attitude hold in the pitch axis. Additionally, a yaw accelerometer is installed to provide a turn coordination feature through a washout circuit of the pitch and yaw SAS. The influence of SAS control from the pilot control inputs (open loop) is controlled by the ASE computer, keeping the SAS 10-percent authority balanced through the integration of signal inputs from the individual LVDTs. Small cyclic and pedal control movements essentially become gain inputs to the SAS servo actuators, allowing full SAS authority. For larger or rapid cyclic control inputs, the SAS rate to the servo actuators receives a momentary increase, which then is washed out once the rate has been established. The purpose of this is to provide the pilot with a command augmentation system, giving the pilot a cyclic control "quickening" with the SAS engaged (fig. 10). Although the quickening circuit is provided for both the roll and pitch axis, the roll or lateral input rates are essentially the same as the basic SAS without quickening in order to help improve control harmony.

Control Programming Unit

14. The control programming unit uses several variable inputs in controlling the longitudinal FFS and wing flaps. The FFS (fig. 10) provides the pilot with a stick force per g during maneuvering flight, and an increased stick force gradient over the basic trim system. The FFS receives inputs from the cyclic longitudinal LVDT, airspeed, and pitch rate gyro. An FFS servo hardover generates approximately 20 pounds push/pull on the cyclic. The FFS is controlled by a switch located on the pilot ASE control panel.

Wing Flaps

15. The wing flaps are variably controlled airfoils designed to increase maneuverability under g loading. They are programmed to reduce airflow restrictions during autorotations. Operating simultaneously, the flaps are automatically positioned from a preset schedule presented in figure 11.

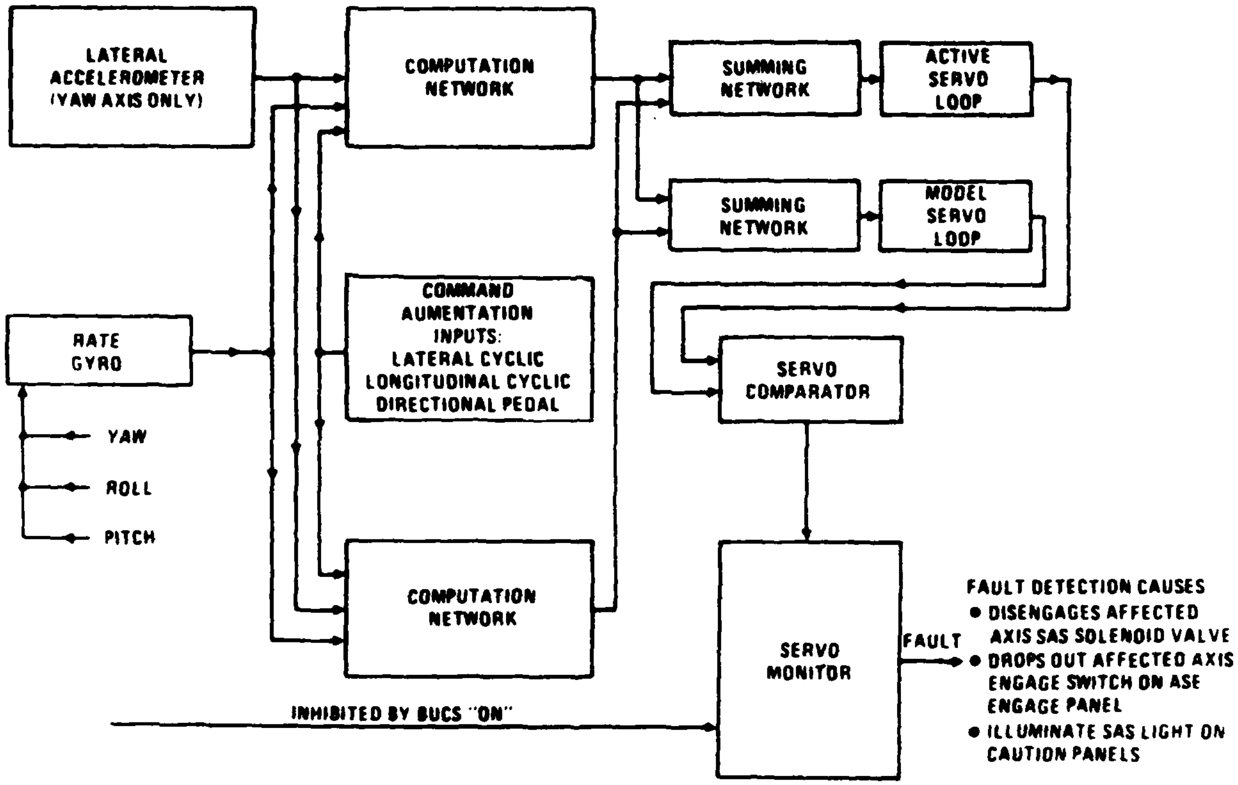


Figure 9. In Flight Monitoring Built-in Test Block Schematic.

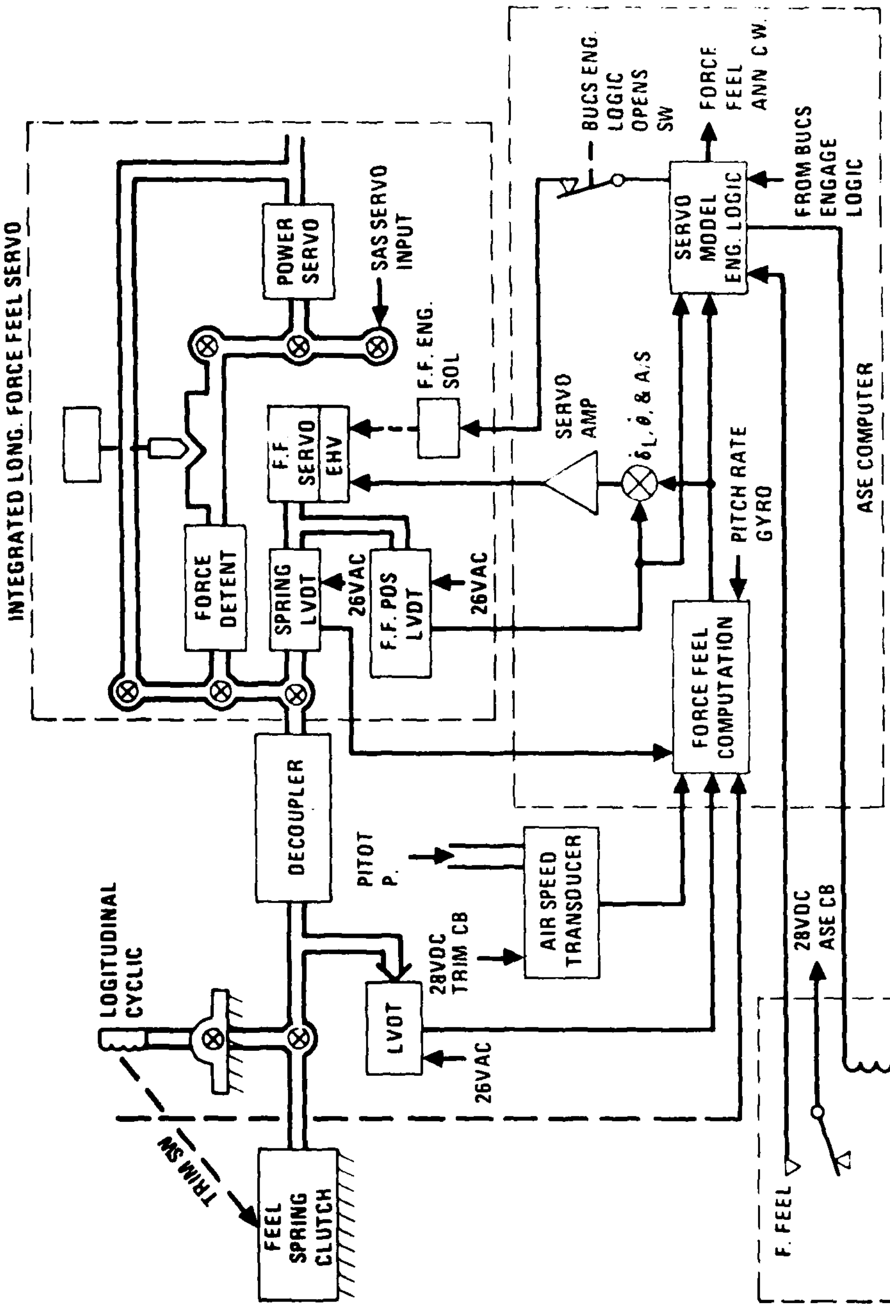


Figure 10. Force Feel System Schematic.

Case	Crew Station Controls	Other Conditions	Flap Position
1	Coll. stick: Full down (auto rotation)	---	45° Up
2	Coll. stick: Greater than 5.2 inches	Airspeed less than 100 knots	20° Down
3	Coll. stick: Greater than 4.0 inches longitudinal cyclic stick inputs	Airspeed greater than 100 knots	5° Down plus inputs to assist pitch control with show washout to 5° down
4	Action switch: Rocket position	Airspeed less than 50 knots pods 10° nose down	25° Up
5	Action switch: Rocket	Airspeed greater than 50 knots pods 10° nose down	5° Down

Notes: (A) Case 1 overrides cases 4 & 5
(B) Case 4 overrides case 2
(C) Case 5 overrides cases 2 & 3

Figure 11. Wing Flap Position Chart.

APPENDIX D. ENGINE DESCRIPTION

GENERAL

1. The primary power plant for the YAH-64 helicopter is the General Electric YT700-GE-700 front drive turboshaft engine, rated at 1536 shp (sea level, standard day, uninstalled). The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: a cold section, a hot section, a power turbine, and an accessory section. Design features of each engine include an axial-centrifugal flow compressor, a through-flow combustor, a two-stage air-cooled high-pressure gas generator turbine, a two-stage uncooled power turbine, and self-contained lubrication and electrical systems. In order to reduce sand and dust erosion and foreign object damage (FOD), an integral particle separator operates when the engine is running. The YT700-GE-700 engine also incorporates a history recorder which records total engine events. Pertinent engine data are shown below.

Model	YT700-GE-700
Type	Turboshaft
Rated power (intermediate)	1536 shp, sea-level, standard-day, uninstalled
Output speed (Np 100%)	20,000 rpm
Compressor	5 axial stages, 1 centrifugal stage
Variable geometry	Inlet guide vanes, stages 1 and 2 stator vanes
Combustion chamber	Single annular chamber with axial flow
Gas generator turbine stages	2
Power turbine stages	2
Direction of rotation (aft looking fwd)	Clockwise
Weight (dry)	415 lb (max)
Length	47 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624, JP-4 or JP-5
Lubricating oil	MIL-L-7808 or MIL-L-23699
Electrical power requirements for history recorder and Np overspeed protection	40W, 115VAC, 400 Hz
Electrical power requirements for anti-ice valve, filter bypass indication, oil filter bypass indication, and magnetic chip detector	1 amp, 28VDC

ENGINE MODULES

2. The engine (fig. 1) consists of four separate modules which are described in the following subparagraphs.

a. The cold section module includes an inertial inlet particle separator incorporating an engine-driven blower mounted on the accessory gearbox. This module also includes the transonic six-stage compressor and the output shaft assembly which interfaces with the helicopter transmission shaft. The compressor has five axial stages and one centrifugal stage. The axial section is transonic, with variable inlet guide vanes and variable first- and second-stage stator vanes. Operation of the compressor variable geometry components is discussed in paragraph 10.

b. The hot section module contains an axial-type annular combustor. The combustor liner is cooled by air impingement and air film. The two-stage gas generator turbine assembly is also included in the hot section module.

c. The power turbine module includes the power turbine, exhaust frame, the shaft, and sump assembly. The power turbine rotor has two stages with uncooled, shrouded tips. The power turbine shaft rotates inside of the gas generator rotor shaft and extends to the front of the engine. The power turbine shaft contains a torque sensor tube that mechanically displays the total twist of the shaft. A diagram of the engine torque system is shown in figure 2. A concentric reference shaft is secured by a pin at the front end of the power turbine drive shaft and is free to rotate relative to the power turbine drive shaft at the rear end. The relative rotation is due to transmitted torque and the phase angle between the reference teeth on the two shafts is picked up by the torque/overspeed sensor. Power turbine speed (Np) is also picked up from these teeth by the Np sensor, which is mounted in the same location.

d. The accessory section module includes the top-mounted accessory drive gearbox and a number of line replaceable units (LRU). An LRU is an item authorized to be removed and replaced with an interchangeable item. The LRU mounted on the aft side of the accessory gearbox includes the HMU, sequence valve, particle separator blower, and engine starter. The LRU mounted on the forward side includes the fuel filter, lubricating oil cooler, fuel boost pump, chip detector, lubricating and scavenge pump, bypass sensor, and alternator stator. The housing of the engine lubricating and scavenge pump and cored passages for oil and fuel are integrally cast into the gearbox, reducing external lines and fittings.

ENGINE SUBSYSTEMS

3. The engine has four basic subsystems: lubrication, fuel, electrical, and air. Other subsystems of the engine include the variable geometry linkage assembly and internal washing system.

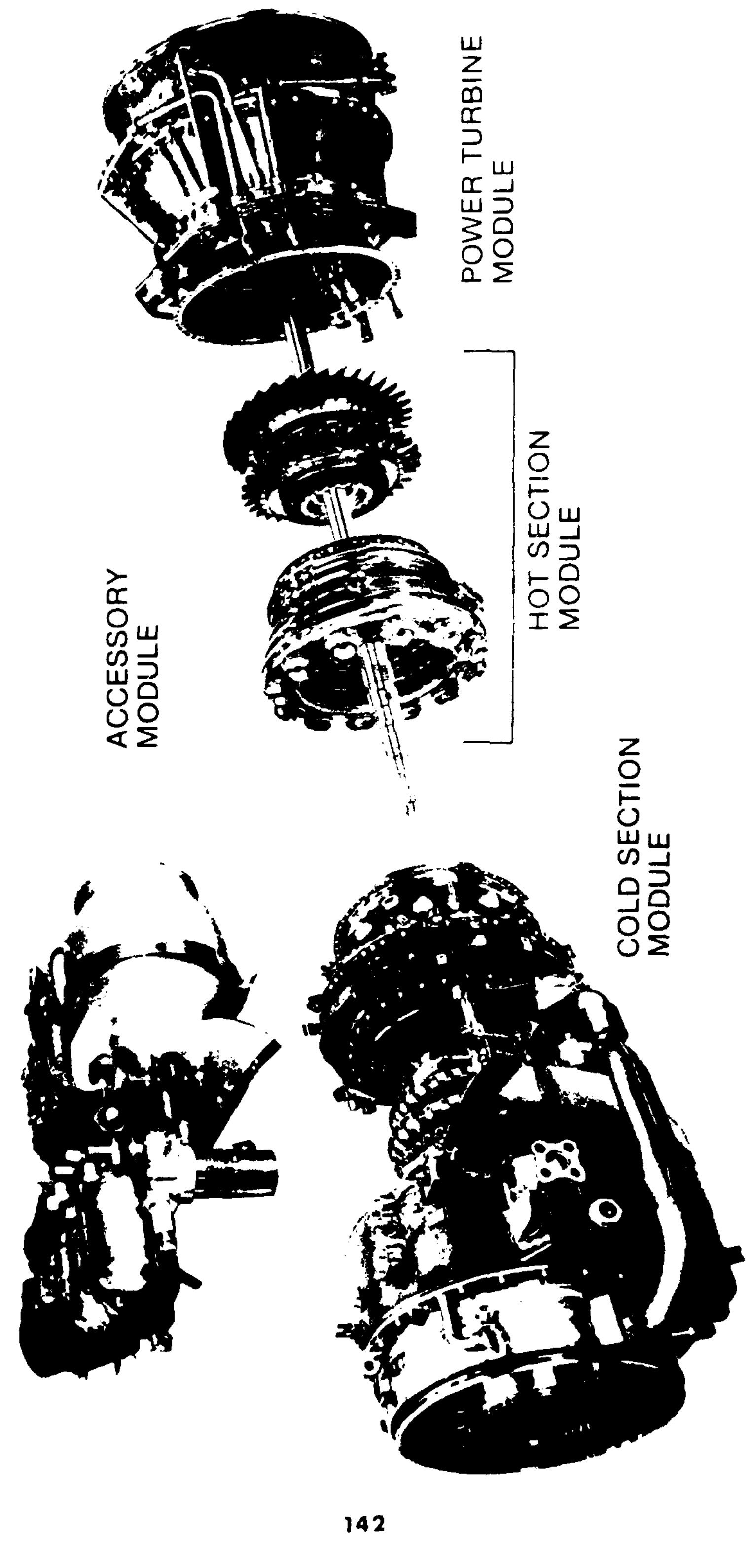


Figure 1. YT-700 GE-700 Engine.

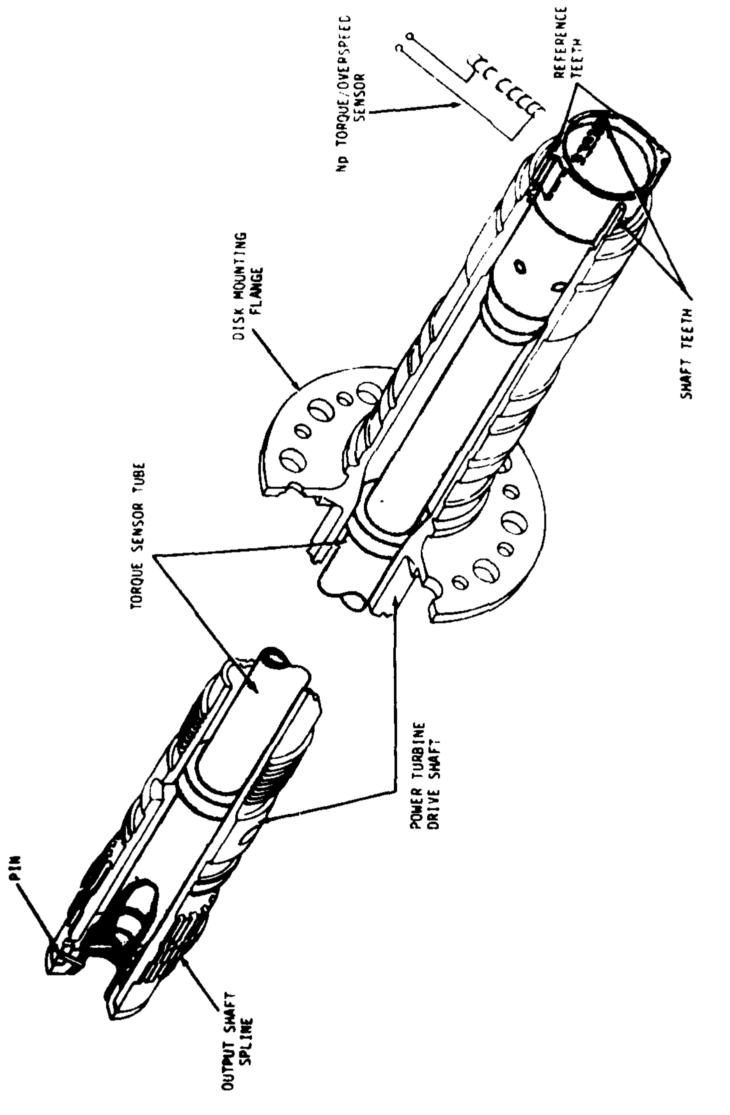


Figure 2. Engine Torque Sensor.

Lubrication System

4. The lubrication system is a self-contained, pressurized, recirculating dry sump system. The system consists of an integral oil tank, lubricating supply and scavenge pump, scavenge screens, oil filter, oil filter impending bypass and indicator, oil filter bypass valve and switch, chip detector, oil sampling port, sight gages, gravity-fill port with screen, oil cooler, pump cold-starting relief valve, emergency oil reservoirs, and sump distribution systems. The system is capable of supplying and scavenging oil, emergency bearing lubrication, and filtering and monitoring the condition of the oil. A schematic of the lubrication system and bearing and sump location is shown in figure 3. Oil from the supply tank is pumped through the filter and through cored passages in the accessory gearbox, where the flow divides to the A, B, and C sumps in the engine and the accessory drive gearbox. Scavenged oil flows through the scavenge screens, chip detector, the fuel-oil cooler, the engine inlet scroll vanes, and returns to the supply tank. An emergency system provides oil mist to lubricate the bearings if the primary oil system fails (fig. 4). Small integral oil reservoirs, located in each sump, are kept full during normal operation by the oil pump. Oil from these reservoirs passes through the oil mist nozzles to provide at least 6 minutes of lubrication.

Fuel System

5. The fuel system consists of the engine-driven boost pump, filter, HMU, sequence valve, primer and main fuel manifolds, primer nozzles, and main fuel injectors. A schematic of the fuel system is presented in figure 5. Fuel from the tank passes through the engine-driven boost pump a reusable filter, and the HMU high-pressure pump. High-pressure fuel is diverted to the wash filter, which supplies finely filtered fuel for the HMU computing servos. The metering pressure regulator valve and the metering valve respond to a signal from the HMU to schedule the required amount of fuel to the engine. The fuel not diverted to the HMU or through the high-pressure bypass valve flows through a metering valve (controlled by the HMU), through a shutoff valve, a pressurizing valve, and the oil cooler to a sequence valve. The sequence valve has four functions. First, it schedules fuel to the primer nozzles and main fuel injectors for starting and engine operation. Second, it purges fuel from the primer nozzles by directing compressor discharge (P_3) air through them after primer nozzle shutoff; this prevents coking of the nozzles. Third, it drains fuel from the main fuel manifold on shutdown to prevent coking. Fourth, it has a bypass valve for power turbine overspeed protection. To accomplish the fourth function the sequence valve contains a solenoid valve which is actuated by a power turbine overspeed signal controlled by the ECU. Operation of the solenoid valve causes some of the fuel coming from the HMU to be diverted from the combustion section back to the inlet of the HMU. The fuel flow to the combustor is transiently reduced to a level which reduces power turbine speed to prevent destructive overspeed. Once power turbine speed is reduced below the ECU overspeed reference valve, the signal to the solenoid ceases and the engine control system governs power turbine speed normally. The overspeed system also contains a cockpit test function (fig. 6) which permits the system to be checked while the engine is running.

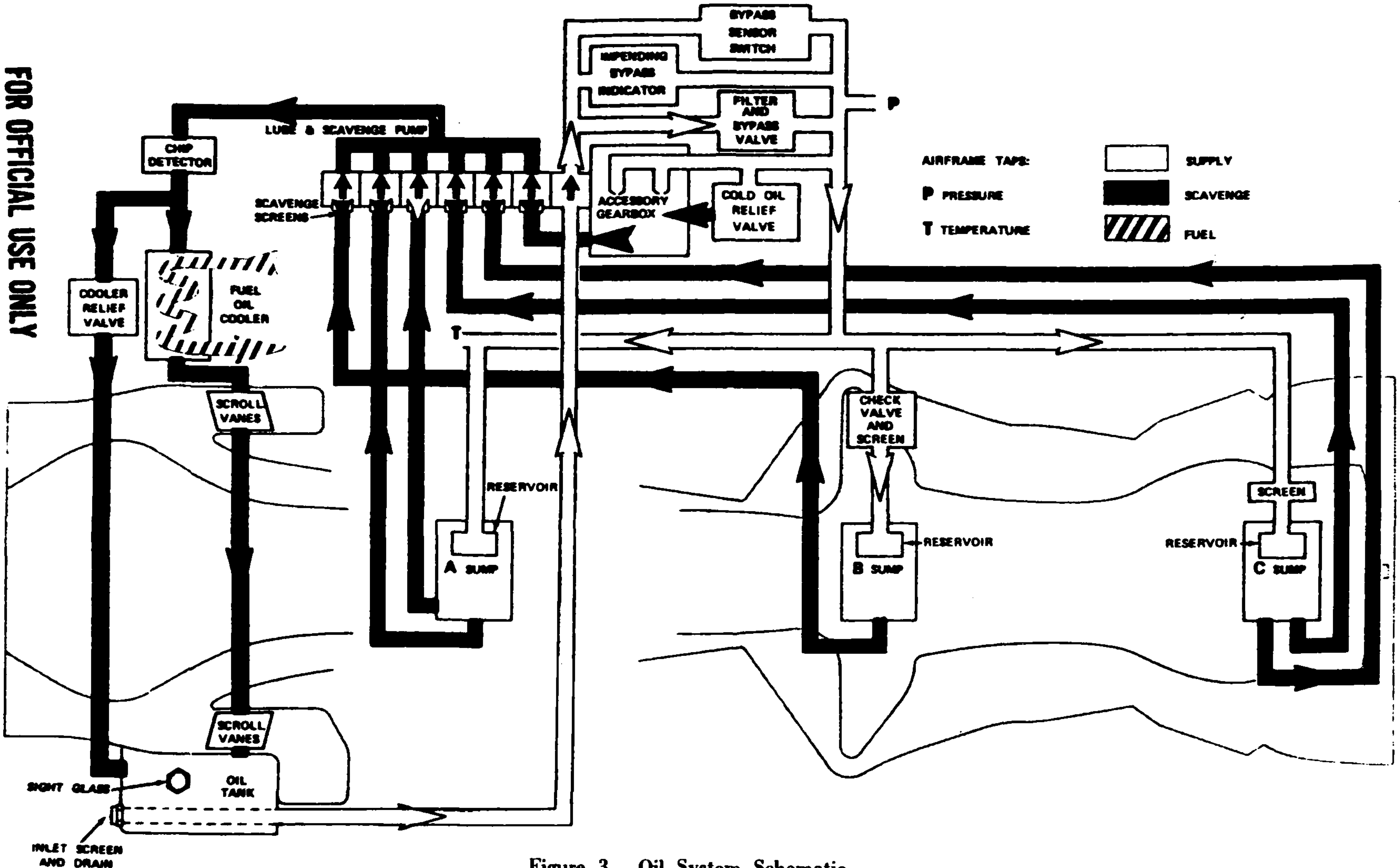


Figure 3. Oil System Schematic.

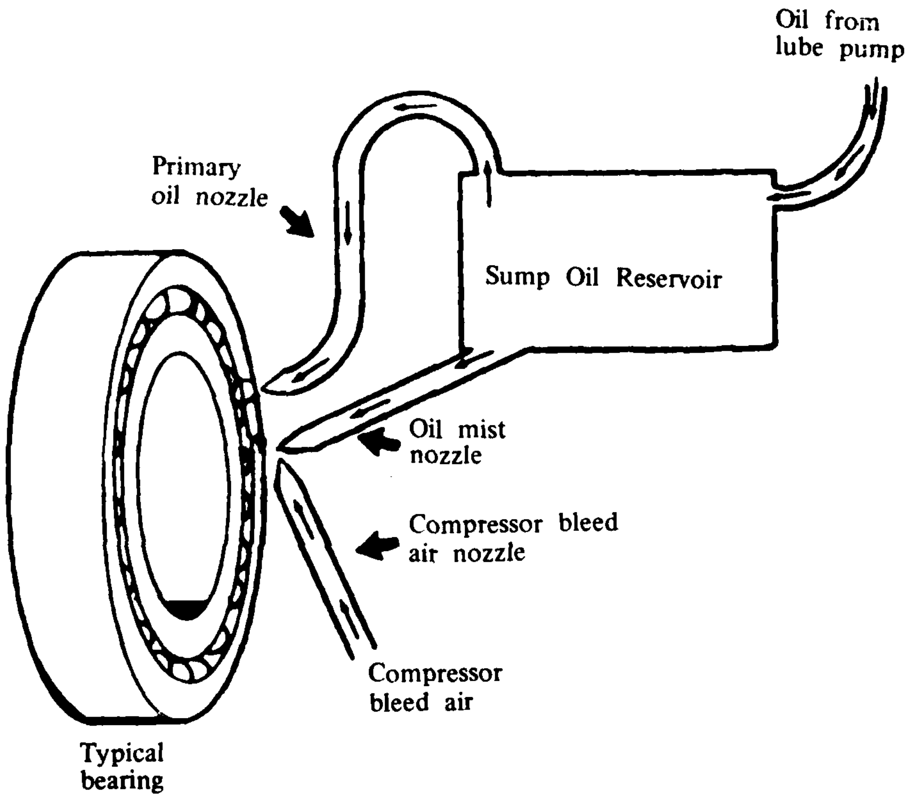


Figure 4. Emergency Oil System.

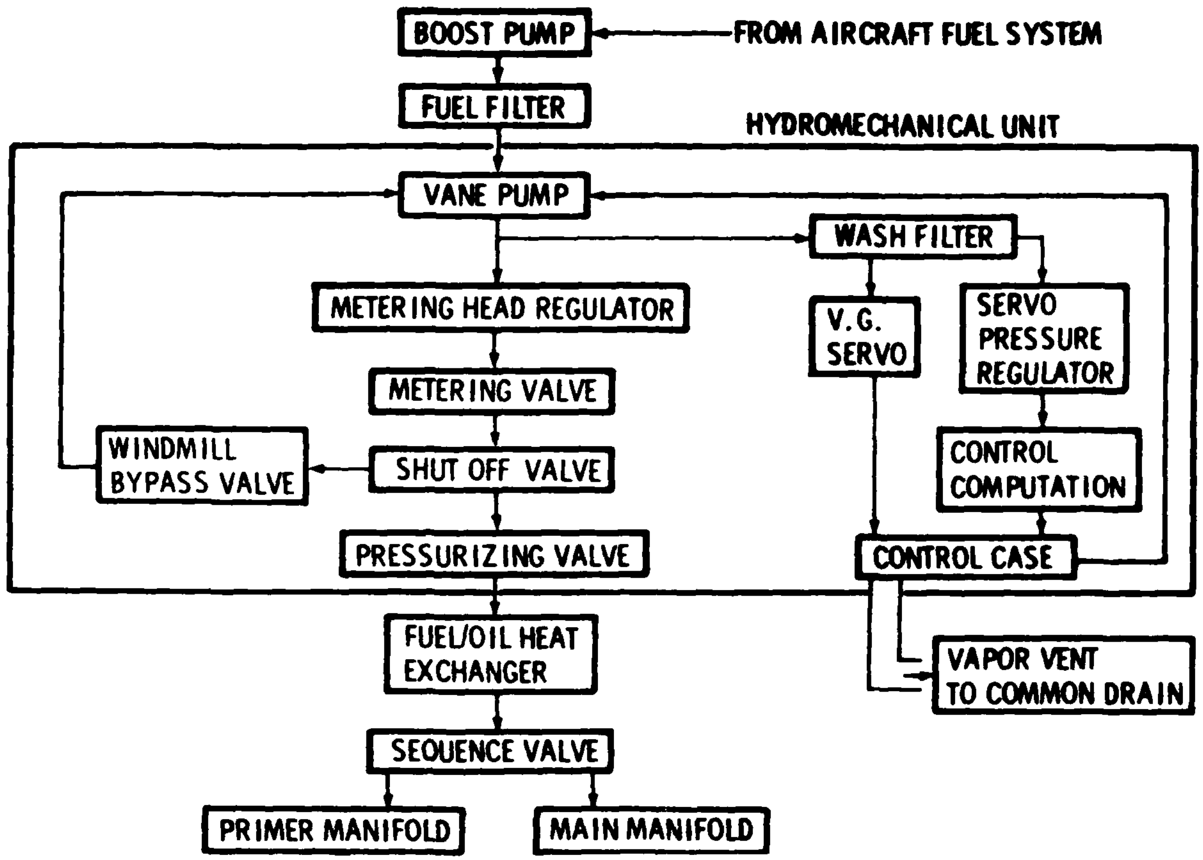


Figure 5. Fuel System Schematic.

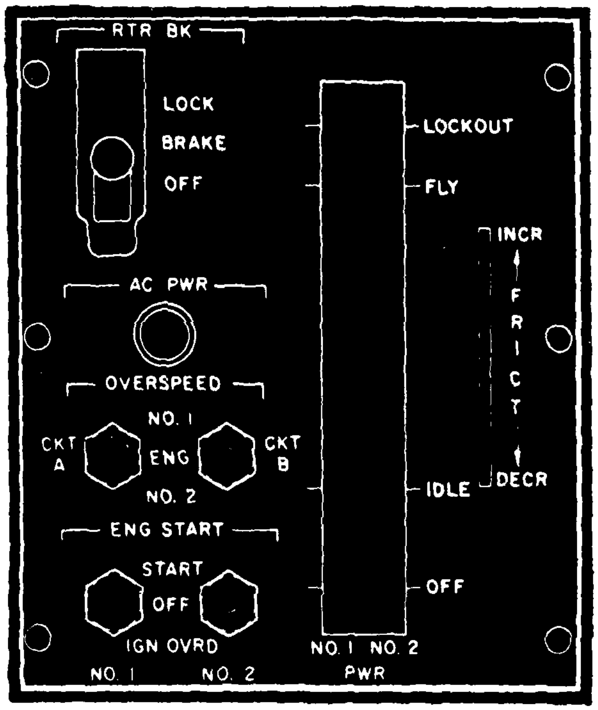


Figure 6. Cockpit Overspeed Control.

6. Control of fuel to the combustion system is accomplished by the HMU. The HMU contains a vane-type high-pressure pump and a variable geometry actuator. The HMU receives two separate linkage inputs from the cockpit. One input is provided by a linkage connected from the power control lever (PCL) to the power-available spindle (PAS) on the HMU. The second input comes from the collective to the load demand spindle (LDS). The ECU provides a signal to the HMU which provides for constant Np over the full power range. The HMU also responds to inputs from compressor inlet temperature (T_2) and compressor discharge pressure (P_3).

Electrical System

7. The engine electrical system has five components: the ECU, alternator, ignition system, $T_{4.5}$ thermocouple harness, and Np and torque/overspeed sensors. Their functions are described in the following subparagraphs.

a. The ECU provides engine control functions, conditioned signals for the engine history recorder and cockpit indications, and test points to a ground connector for electrical and engine system diagnostics. The following control functions are provided: constant Np governing to within ±1 percent of sensed Np; $T_{4.5}$ temperature limiting with an accuracy of ±5°C; Np overspeed protection completely independent of the normal Np governor; and load sharing on torque to within ±5 percent of IRP torque. It also provides the following noncontrol signals: Np to the cockpit; torque signal to the cockpit; $T_{4.5}$ signal to the cockpit; $T_{4.5}$ signal to the history recorder for overtemperature events and time-temperature integration; $T_{4.5}$ signal to ground units for engine diagnostics; and DC power and 400-Hz power to the history recorder.

b. The alternator is gearbox-mounted and has three separate windings. Winding No. 1 supplies the ignition exciter. Winding No. 2 supplies power to the ECU and to the primary Np overspeed circuit. Winding No. 3 supplies the N_G cockpit signal.

c. The ignition system is a noncontinuous AC-powered capacitor discharge system. It includes an ignition exciter, two igniter plugs, and two ignition leads. The ignition duty cycle is 2 minutes ON, 3 minutes OFF, 2 minutes ON, and 23 minutes OFF.

d. The $T_{4.5}$ thermocouple harness is a five-probe dual-immersion harness using chromel-alumel functions to provide signals to the ECU for $T_{4.5}$ limiting and cockpit indication. Each of the five probes is individually wired to a multipin connector, allowing diagnostic checks to be made for open or grounded elements.

e. The Np, Np overspeed, and torque sensing are provided by two identical variable reluctance pickups. The Np sensor provides a basic signal to the ECU. The torque and overspeed sensor senses power turbine torque and provides a speed signal to the separate Np overspeed protection system.

8. The electrical system components are engine-mounted and self-contained. The power sources and the components they power are as follows:

a. Alternator winding No. 1 - Ignition exciter.

b. Alternator winding No. 2 - Electrical control unit and primary Np overspeed circuit.

c. Alternator winding No. 3 - N_G cockpit signal.

d. Airframe 400-Hz, 115-VAC - History recorder and backup Np overspeed circuit.

e. Airframe 28-VDC - Anti-icing valve, oil filter bypass, fuel filters bypass, and magnetic chip detector.

Air System

9. The air system performs the following functions: cools the turbine section and provides anti-icing air, seal pressurization, sump venting, airframe bleed air requirements, and compressor discharge pressure (P_3) signal to the HMU. These functions are described in the following subparagraphs.

a. Compressor discharge leakage air is used to cool the stage 1 and stage 2 nozzles. Air leaking from the centrifugal compressor at the diffuser is ducted through the mid frame and into the nozzle vanes. The air cools the vanes and exits through the holes in the vane airfoils.

b. Anti-icing is achieved by a combination of controlled (ON/OFF) usage of axial compressor discharge bleed air and continuous heat rejection from the air-oil cooler, which is an integral part of the scroll vanes. The compressor discharge bleed air anti-ices the swirl frame and swirl vanes. Control of anti-icing air is provided by a solenoid-operated anti-icing valve which is actuated by a cockpit switch. The switch is fail-safe, in that when electrical power is supplied to the anti-ice valve solenoid, the anti-icing air is turned off. When power is off the valve is open.

c. Seal pressurization prevents oil loss from sumps by controlled airflow. It prevents hot gases, dust, and moisture from entering sumps and provides air for the emergency oil system.

VARIABLE GEOMETRY LINKAGE ASSEMBLY

10. The compressor variable geometry components consist of a fuel-driven actuator integral with the HMU; a crankshaft with the necessary links to attach to the actuating rings of the inlet guide vanes; first- and second-stage variable vanes; and the anti-icing and start bleed valve. Rotation of the crankshaft by the HMU

actuator translates to circumferential movement of the actuator rings, which results in synchronized opening or closing of the variable vanes and opening or closing of the anti-icing and start bleed valve.

ENGINE INTERNAL WASHING SYSTEM

11. The engine wash manifold, integral with the swirl frame, has jets aimed at the compressor inlet annulus in the front frame. The wash manifold fitting is located at the 6 o'clock position on the swirl frame.

PNEUMATIC START SYSTEM

12. The pneumatic start system uses an air turbine-type engine start motor. System components include the shaft-driven compressor (SDC), left engine bleed air valve, engine start motor, start control valve, external start connector and check valve, controls, and ducting. A schematic of the bleed air control system is presented in figure 7. Three air sources may provide air for engine starts: the SDC, left engine bleed air, or ground air source. Starts are accomplished in part through an electrically operated start control valve that provides regulated air flow to the pneumatic start system. Pressure regulation prevents an overtorque situation when starts are conducted at high bleed air pressures. The start control valve is designed so that downstream pressure builds gradually to prevent impact damage of the engine starter pad. The pneumatic starter turbine wheel drives the engine through a gearbox and an overrunning clutch. A starter speed switch wired to the start control valve terminates the start cycle when cutoff speed is reached. The external start connector is on the left side of the fuselage. It is the attachment point for a bleed air line from an external source for engine starting. The connector contains a check valve to prevent engine or SDC bleed air from being vented.

ENGINE CONTROL FEATURES

Cockpit Engine Controls

13. The engine controls provide for the more common functions of engine fuel scheduling, variable guide vane geometry control, power modulation for rotor speed control, and overspeed protection. The system also incorporates control features for torque matching and overtemperature protection. Fully integrated operation of the engine results from three primary crew station inputs. A special control feature is provided which quickly reduces engine power during certain emergencies.

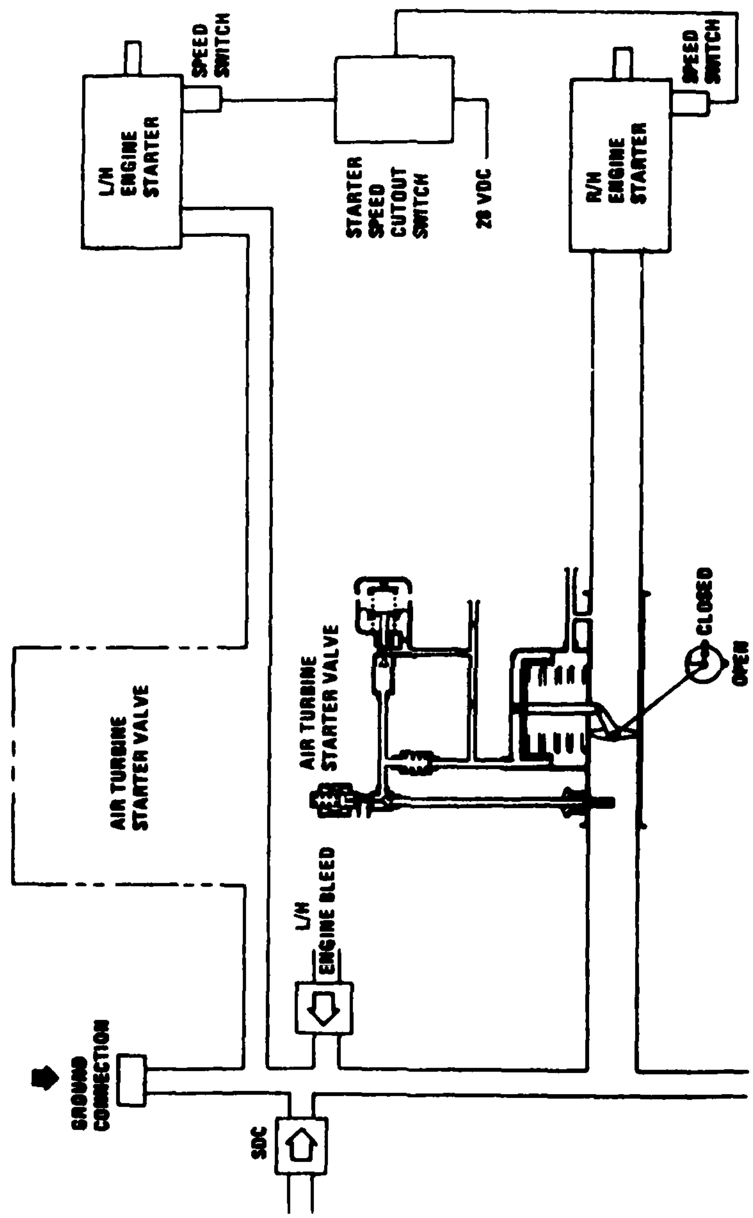


Figure 7. Pneumatic Schematic.

Power-Available Spindle and Power Control Levers

14. The PAS, located on the HMU, mechanically stopcocks fuel and sets permissible N_G for operating conditions from ground-idle through maximum available power. Two PCL's, which set the position of the PAS, are located on the engine power quadrants in each crew station. Markings on the quadrant identify which engine the PCL controls (No. 1 or No. 2). Position markings are OFF, IDLE, FLY, and LOCKOUT. In the OFF position fuel to the engine is stopcocked. In the IDLE position, the gas generator operates at ground-idle speed. During normal operations the PCL's are advanced to the FLY position where the engines are automatically controlled. Further advancement of the levers to the LOCKOUT position disables the automatic feature by interrupting ECU commands to the HMU, thus allowing manual control of the engines. In order to reset automatic control the levers must first be retarded to IDLE, then advanced to FLY. Mechanical stops on the pilot quadrant prevent the PCL's from being moved from FLY to LOCKOUT, or from IDLE to OFF without activating the stop release on the PCL's. The stop releases on the copilot/gunner PCL's operate electrically while those on the pilot PCL's are mechanical. A friction control is provided on the pilot power quadrant.

Load Demand Spindle

15. The LDS, also located on the HMU, is linked to the collective sticks. The collective provides an input to the LDS as a function of collective pitch. This signal results in engine compensation to reduce rotor transient droop.

Power Turbine Trim Switch

16. The power turbine trim (beep) switch, located on the collective sticks, is used to provide an electrical signal to the ECU for trimming power turbine speed. The trim range is from 94 to 104 percent N_P, with a normal operating range of 98 to 100 percent N_P. Figure 3, appendix B, illustrates the collective stick grip.

Engine-Cut Switch

17. The engine-cut switch is a recessed pushbutton located on the collective stick grip in each crew station. Activation of this switch automatically calls for a reduction of engine power on both engines to idle. Reactivating the switch returns engine speed to its previously governed value. When the engine-cut circuit is in operation the ENGS CUT caution light illuminates and the ENGS CHOP warning light flashes. When activated, the engine-cut circuit normally produces a split between N_P and N_R, if the collective is low and the aircraft is not hovering. If N_P and N_R do not split, then engine power should not be restored by pressing the ENG CUT switch a second time without first retarding both PCL's to IDLE. Once the PCL's are retarded to IDLE and the ENG CUT switch pressed to restore engine power, the PCL's may then be advanced to FLY.

Automatic Governing Characteristics

18. In general, the HMU provides for gas generator control in the areas of acceleration limiting, stall and flameout protection, N_G control, rapid response to power demand, and variable geometry actuation. The electrical unit trims the HMU to satisfy the requirements of the load so as to maintain rotor speed and load sharing and also to limit engine turbine inlet temperature. The basic control and governing functions of these two units are outlined in table 1 and schematically shown in figures 8 and 9.

19. Control of the gas generator is by a droop N_G governor in the HMU. The N_G governor reference is set by the PAS angle and modified mechanically by the LDS angle and trimmed electrically by an input from the ECU through an electrical-mechanical interface device called a "torque motor." After the PAS is set at 120 degrees with the PCL's, the load demand signal is then provided through the LDS. As the LDS is reduced from its maximum setting with a reduction of collective pitch, the desired N_G is reset down from the prevailing PAS schedule to provide a gas generator response. This reset schedule is then trimmed by the ECU to satisfy the Np and load control functions established by the ECU. This results in a zero steady-state Np error. The electrical trimming signal is a result of the Np governor, load-sharing circuit, and $T_{4.5}$ limiter, as combined in the ECU. The signal causes a resetting of the "collective compensation" curve, as shown in figure 10. In response to the resulting N_G reference, the HMU operates as a conventional gas generator power control and reschedules fuel flow within preset limits to obtain a reference N_G.

Manual Governing Characteristics

20. Failure of the ECU causes automatic Np governing to become inoperative. The Np overspeed system and LDS reset still remain operable. During this failure mode, the engine is controlled by use of the PAS and LDS. Without an electrical input, the HMU reverts to a power control. This power control is a droop control of N_G to a value called for by the PAS position as reset by the LDS input. The PAS position is set by manually adjusting the PCL's.

21. In the event the ECU torque motor fails to a lower engine power, the torque motor can be mechanically deactivated. By advancing the PAS past intermediate to 130 degrees, the ECU interface is deactivated and locked out. Engine power is then controlled through the PAS by manually adjusting the PCL's. Since this deactivation of the ECU interface is at the HMU torque motor, it does not affect any other ECU functions such as torque computation, Np overspeed protection, or signal generation. It does deactivate Np governing, $T_{4.5}$ limiting, and load sharing, which all normally act through the torque motor.

Table 1. Fuel and Control System Functional Split.

HYDROMECHANICAL UNIT

- Fuel pumping
- Fuel flow metering
- Collective pitch compensation through LDS
- Acceleration and deceleration flow limiting (including starting)
- N_G limiting
- Variable geometry positioning
- Torque motor to trim N_G governor
- PAS override and control with electrical unit inoperative
- Vapor vent on PAS overtravel for fuel system priming

ELECTRICAL CONTROL UNIT

- HMU trimming of N_G governor as determined by:
  a. Isochronous N_P governing
  b. $T_{4.5}$ limiting
  c. Load sharing on torque
  d. N_P reference input from cockpit
- Redundant N_P overspeed limit
- Cockpit signal generation of N_P, $T_{4.5}$ and torque
- History recorder signals

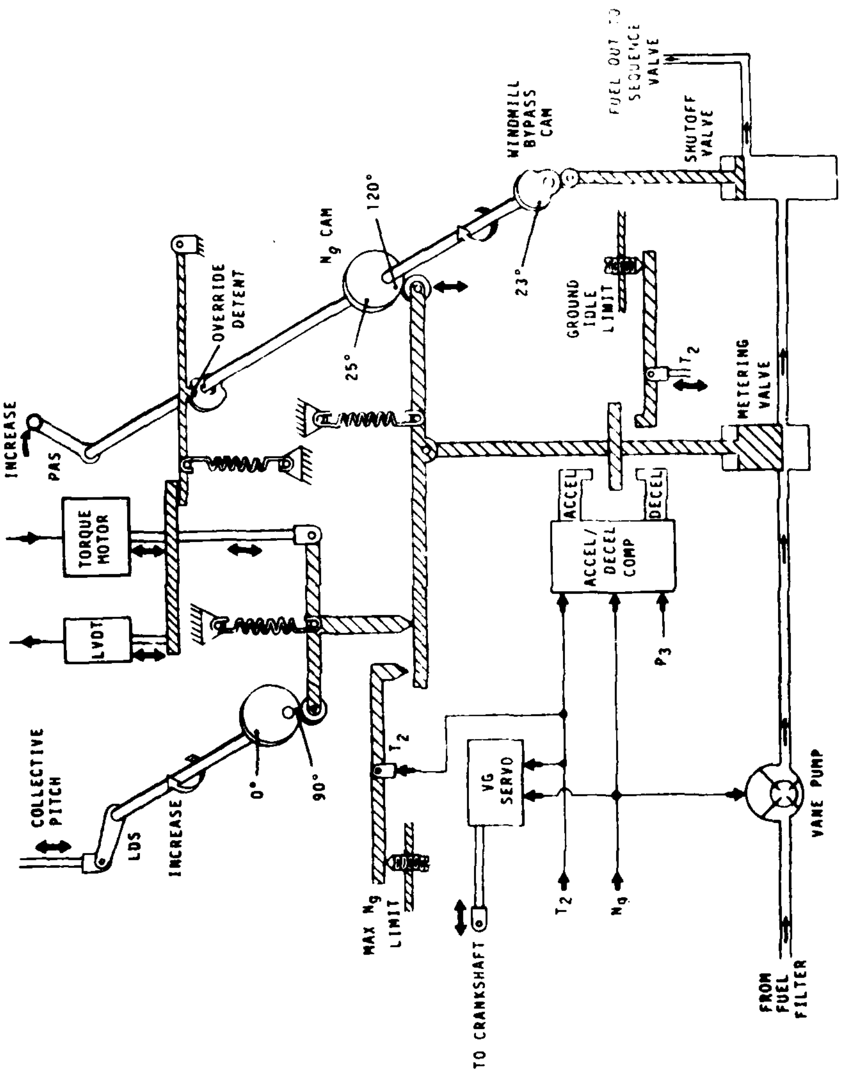


Figure 8. HMU Schematic Diagram.

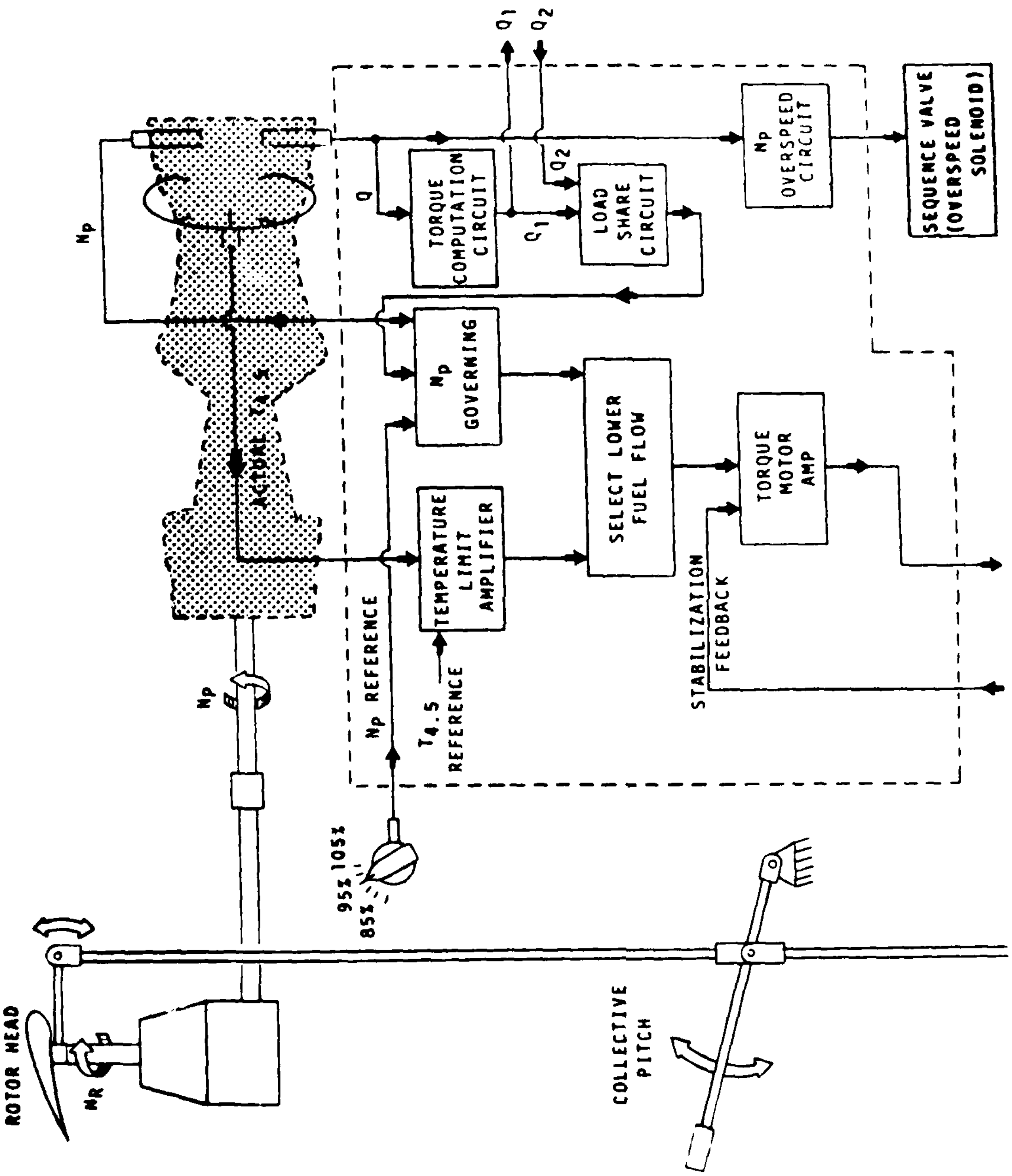


Figure 9. ECU Schematic Diagram.

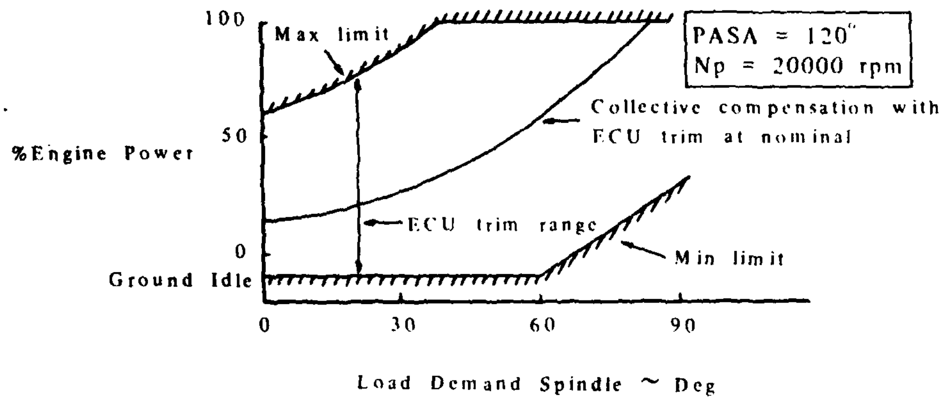


Figure 10. Collective Compensation Curve.

AUXILIARY POWER UNIT

22. The International Harvester-built Model T62T-40-4 APU consists of a gas turbine power section, reduction gear drive, and appropriate controls and accessories. The gas turbine power section is a single-spool gas turbine using a single centrifugal compressor and a single-stage radial inflow turbine. The APU is hydraulically started with fluid from an accumulator. Ignition is accomplished by a separate pressure atomizing fuel nozzle and a spark plug. Once ignition and combustion of the main fuel is completed, the spark plug and start fuel are turned off. A purge valve allows compressor discharge pressure to bleed through the start fuel nozzle to continuously keep it clean for the next start. A pinion gear supported by three planetary reduction gears (7.1:1) reduces turbine shaft speed from 61,248 rpm to an output speed of 8216 rpm. The APU is lubricated by an accessory-driven lubricating pump which circulates oil from the gearbox sump to three jets spraying the high-speed pinion gear. The resulting splash and oil mist fills the gearbox cavity and passes through the hollow shaft, lubricating the remaining gears and bearings. Pertinent information concerning the APU is listed below.

Rated engine speed	61,248 rpm
Exhaust gas temperature	1220°F ±10
Weight (dry)	92 lb (approx)
Output shp	90
Fuel consumption at rated power	150 lb/hr (approx)
Reduction gear and accessories:	
Input speed (rated)	61,248 rpm
Output speed (rated)	8216 rpm
Fuel control assembly	4200 rpm
Fuel	JP-4, JP-5
Lubricating oil	MIL-L-7808 or MIL-L-23699
Oil pump pressure at rated speed	15 to 40 psi
Components and systems:	
Compressor	Single-stage, centrifugal flow
Turbine	Single-stage, radial flow
Combustor	Annular type

23. The APU provides power for all aircraft accessories when the rotor system is not turning. The APU drives the accessory gearbox (AGB) through a centrifugal clutch on the APU output shaft and an overrunning clutch which is a part of the AGB. Electrical, hydraulic, and pneumatic accessories are driven through a gear train within the AGB. The APU control panel, mounted in the pilot right console, is illustrated in figure 3, appendix B.

24. An integral automated start sequencer and condition monitor is provided. Once the pilot initiates an APU start the sequencer monitors critical APU parameters both during spin-up and while operating. An N_G indicator is provided on the APU control panel to permit the pilot to monitor APU starts and operation. Should a malfunction be detected, the APU automatically shuts down and the APU FAIL caution light illuminates.

Starting

25.The APU is started by a hydraulic starting motor powered by the utility accumulator. With the rotor system stopped, the accumulator will supply hydraulic power sufficient for one complete APU starting sequence. An APU hang start will normally require recharging the accumulator.

Auxiliary Power Unit Operation

26. The APU is intended for ground operations only. In-flight operation of the APU may create a fire hazard. Therefore, the APU must be shut down prior to takeoff. The APU ON caution light illuminates whenever the APU control switch is in the RUN or START positions to remind the pilot to shut down the APU before flight.

APPENDIX E. INSTRUMENTATION

1. In addition to the standard aircraft instruments, sensitive calibrated instrumentation and a boom extending forward from the nose of the aircraft were installed. The boom incorporated an angle-of-attack sensor, an angle-of-sideslip sensor, and a swiveling pitot-static tube which was used as the basis for calibrated airspeed in this report.

2. The sensitive instrumentation and related special equipment used is listed below.

<u>Pilot Station</u>

Instrumentation controls and lights
Event switch
Time code display

<u>Pilot Panel</u>

Airspeed (boom)
Altitude (boom)
Altitude (radar)[2]
Rate of climb (boom)
Rotor speed
Engine torque[1]
Engine measured gas temperature ($T_{4.5}$)[1]
Engine gas generator speed[1]
Control position:
- Longitudinal
- Lateral
- Directional
- Collective

Center-of-gravity normal acceleration
Angle of sideslip
Low airspeed[2]
Tether cable angle[2] (during tethered hover only)
Collective actuator load
Lateral actuator load
Tail rotor horsepower
Hanger bearing load warning

[1] Both engines.
[2] Performance aircraft only.

Copilot/Engineer Station

Instrumentation controls and lights
Event switch
Control fixtures
Time code display
SAS hardover box

Copilot/Engineer Panel

Airspeed (boom)
Altitude (boom)
Rate of climb (boom)
Rotor speed
Engine torque[1]
Engine measured gas temperature ($T_{4.5}$)[1]
Engine gas generator speed[1]
Free air temperature
Angle of sideslip
Fuel used (totalization)[1]
Time of day
Correlation counter
Tether cable tension[2] (during tethered hover only)
Fuel flow rate[1]

3. Data parameters recorded on board the aircraft are listed below.

Digital (PCM) Data Parameters

Airspeed (ship's system)
Airspeed (boom)
Low airspeed[2]
Altitude (boom)
Altitude (radar)[2]
Free air temperature
Rotor speed
Gas generator speed[1]
Power turbine speed[1]
Fuel used[1]
Fuel temperature[1] (at flowmeters)
Engine fuel flow[1]
Engine output shaft torque[1]
Engine measured gas temperature ($T_{4.5}$)[1]
Engine inlet temperature[1,2]
Engine inlet total pressure[1,2]

[1] Both engines.
[2] Performance aircraft only.

Engine exhaust system static pressure[1,2]
Engine inlet guide vane position[1,2]
Engine compressor discharge static pressure[1,2]
Engine air particle separator discharge static pressure[1,2]
Fuel control discharge fuel static pressure[1,2]
Main rotor shaft torque
Tail rotor shaft torque
Tether cable tension[2]
Time of day (time code generator)
Pilot event
Engineer event
Correlation counter
Control position:
- Longitudinal cyclic
- Lateral syclic
- Collective

Engine condition lever[1]
Wing flap position
Flight control augmentation positions:
- Longitudinal
- Lateral
- Directional

Control forces:
- Longitudinal cyclic
- Lateral cyclic
- Directional

Aircraft attitude:
- Pitch
- Roll

Magnetic heading
Aircraft angular velocity:
- Pitch
- Roll
- Yaw

Aircraft angular acceleration:
- Pitch
- Roll
- Yaw

Acceleration:
- Center-of-gravity longitudinal
- Center-of-gravity lateral
- Center-of-gravity normal

Angle of sideslip
Angle of attack
Main rotor blade flapping angle

[1] Both engines.
[2] Performance aircraft only.

Tail rotor blade angle
Tail rotor teetering angle
Collective and lateral actuator housing loads
Vibration (accelerometers) (aircraft SN 22248):

- Pilot collective vertical
- Pilot collective lateral
- Pilot cyclic vertical
- Pilot cyclic lateral
- Pilot seat vertical
- Pilot seat longitudinal
- Pilot seat lateral
- Pilot heel slide vertical
- Pilot heel slide lateral
- Copilot seat vertical
- Copilot seat lateral
- Copilot seat longitudinal
- Copilot instrument panel vertical
- Copilot instrument panel lateral
- Copilot instrument panel longitudinal
- Pilot instrument panel vertical
- Pilot instrument panel lateral
- Pilot instrument panel longitudinal
- Aircraft cg vertical
- Aircraft cg lateral
- Aircraft cg longitudinal
- Right avionics bay vertical
- Right avionics bay lateral
- Right avionics bay longitudinal
- Left avionics bay vertical
- Left avionics bay lateral
- Left avionics bay longitudinal
- Main transmission vertical
- Tail rotor gearbox vertical
- Left wing vertical
- No. 1 engine exhaust shroud vertical

Vibration (accelerometers) (aircraft SN 22249):

- Aircraft cg vertical
- Aircraft cg lateral
- Pilot panel vertical
- Pilot seat vertical
- Pilot seat lateral
- Copilot seat vertical
- Copilot seat lateral

Analog (FM) Data Parameters

Forward right and left hanger bearing loads

4. Provision was made for telemetry transmission or other real time display of data during these tests.

5. Fixtures to provide for pulse control inputs in the collective, longitudinal, lateral, and directional flight control systems were provided and installed by the contractor.

6. The boom airspeed calibration of the YAH-64 was accomplished by pacing the aircraft with a calibrated AH-1G helicopter. The results of these calibrations are presented in figures 1 and 2.

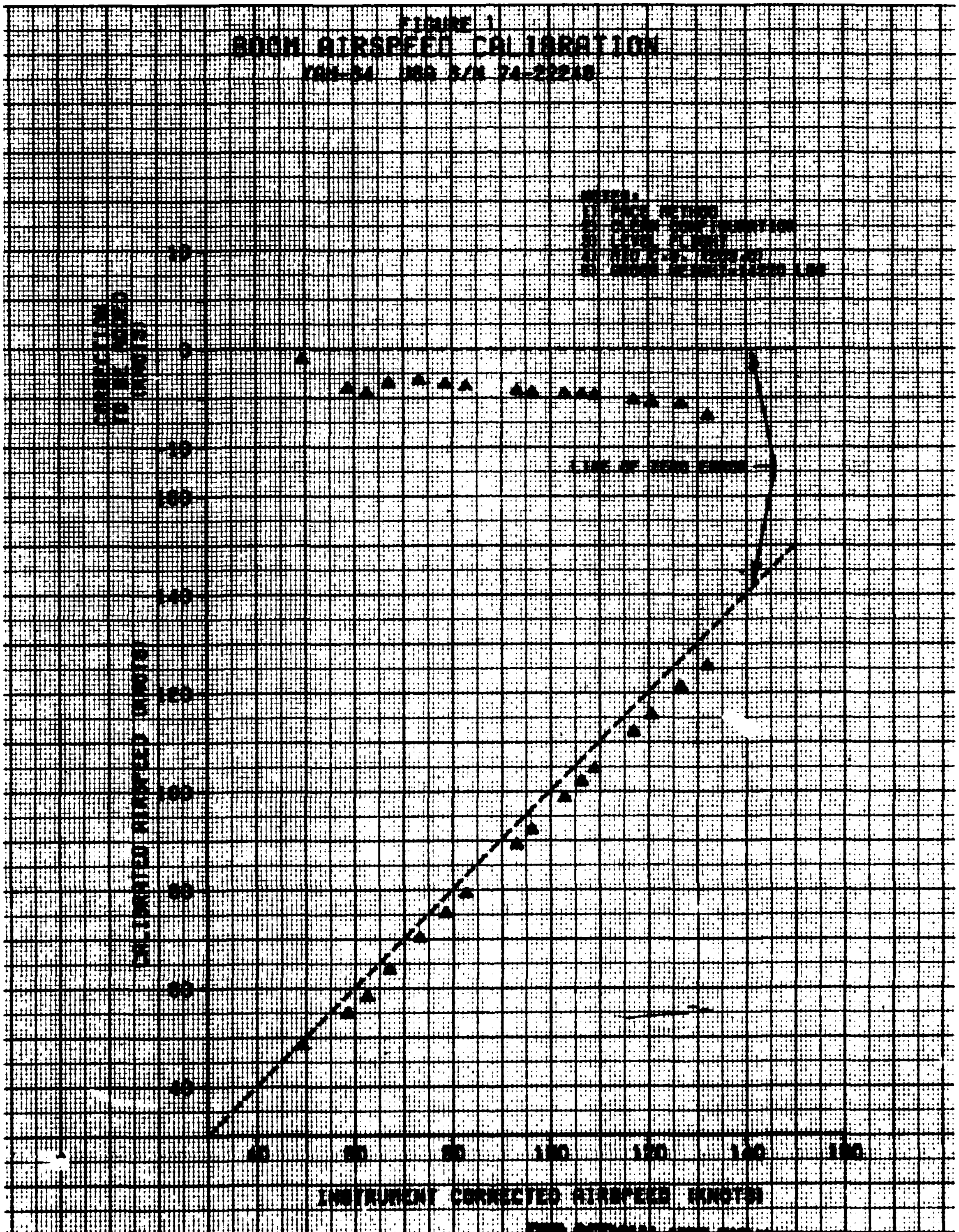


FIGURE 1
BOOM AIRSPEED CALIBRATION
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

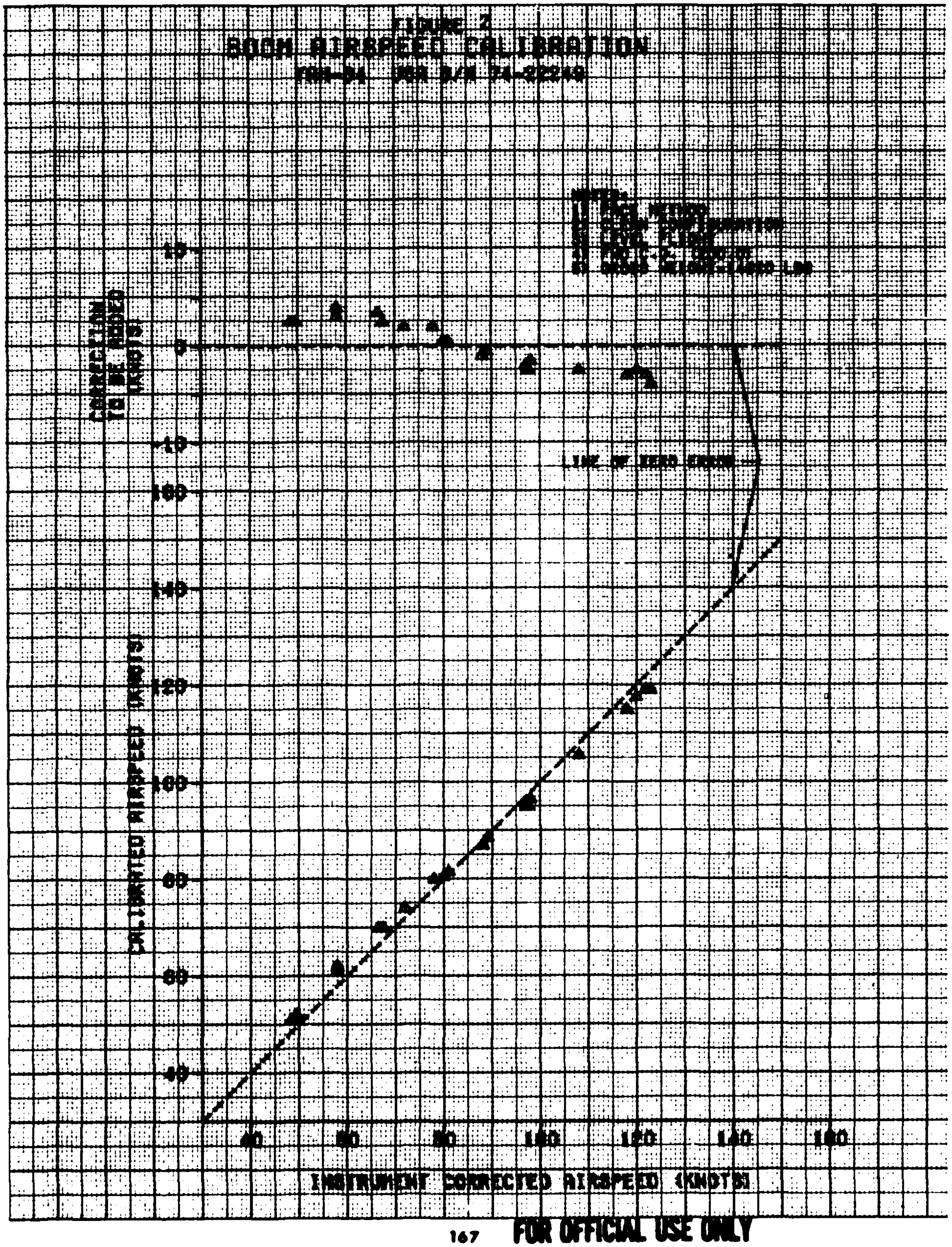


FIGURE 2
BOOM AIRSPEED CALIBRATION
CORRECTION TO BE ADDED (KNOTS)
10
0
-10
LINE OF ZERO ERROR
160
140
120
100
80
60
40
CALIBRATED AIRSPEED (KNOTS)
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

APPENDIX F. TEST TECHNIQUES AND DATA ANALYSIS METHODS

TEST TECHNIQUES

1. Conventional test techniques were used in both the performance and handling qualities tests, with the exception that performance testing was conducted in coordinated (ball-centered) flight instead of zero sideslip. The basic techniques for each test are described in the Results and Discussion section of this report. Detailed descriptions of all test techniques are contained in references 7 and 8, appendix A. The HQRS presented in figure 1 was used to augment pilot comments relative to handling qualities.

DATA ANALYSIS METHODS

General

2. Performance data were evaluated using the methods described in reference 6, appendix A. Handling qualities data were evaluated using standard test methods described in reference 7.

Nondimensional Method

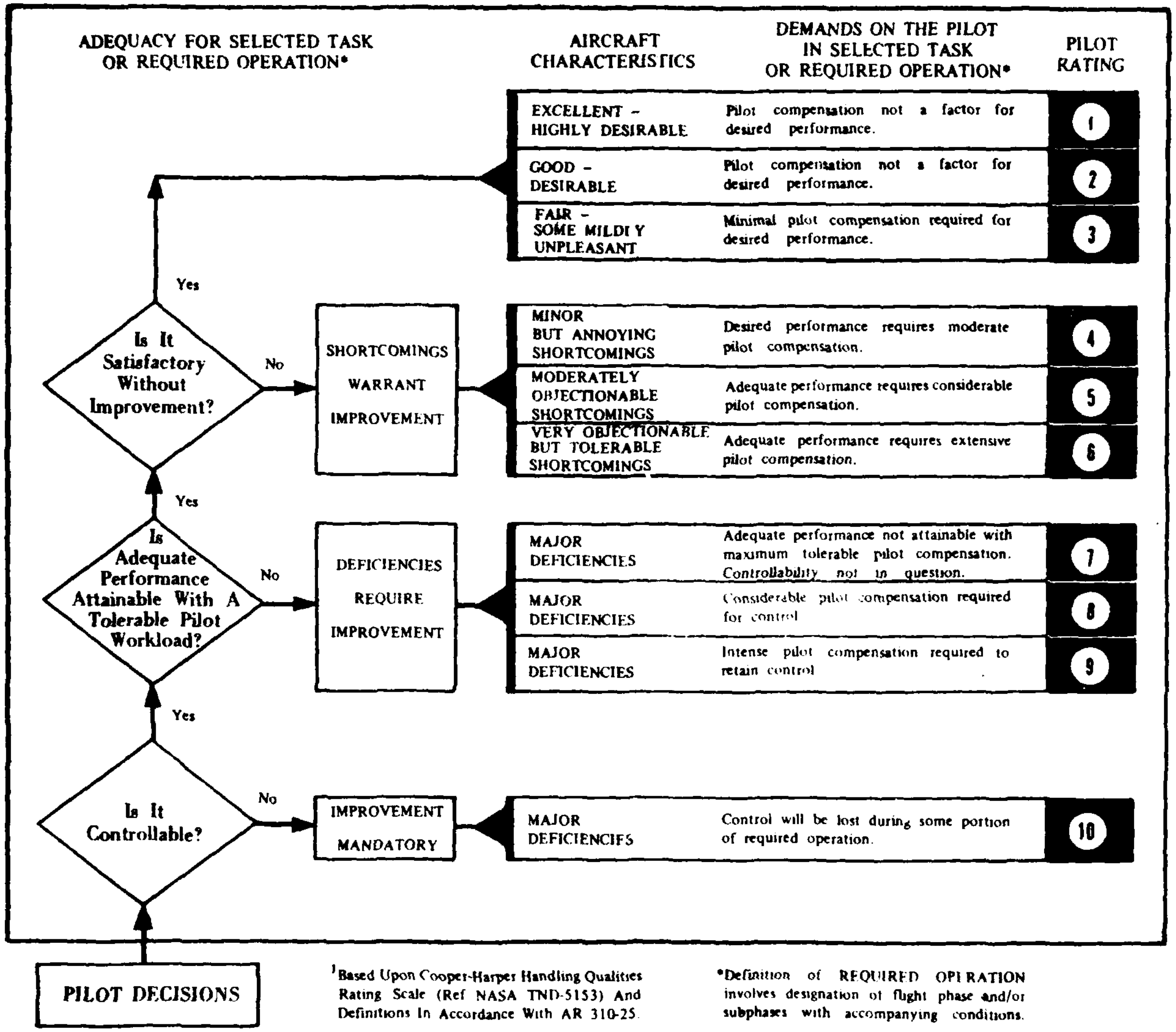
3. Helicopter performance may be generalized through the use of nondimensional coefficients. This method is not valid for separating the effects of retreating blade stall and compressibility. During this evaluation, trends indicating an influence of stall and compressibility were noted. However, it was beyond the scope of this test to determine the effects of these parameters. The following nondimensional coefficients were used:

a. Coefficient of power (C_P):

$$C_p = \frac{shp\ (550)}{\rho A\ (\Omega R)^3} \tag{1}$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{THRUST}{\rho A(\Omega R)^2} \tag{2}$$



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure 1. Handling Qualities Rating Scale.

c. Advance ratio (μ):

$$\mu = \frac{1.6878 \; V_T}{\Omega R} \tag{3}$$

d. Advancing blade tip Mach number (M_{tip}):

$$M_{tip} = \frac{1.6878 \; V_T + (\Omega R)}{a} \tag{4}$$

Where:

SHP = Engine output shaft horsepower (both engines)

ρ = Air density (slug/ft^3)

A = Main rotor disc area (ft^2)

Ω = Main rotor angular velocity (radian/sec)

R = Main rotor radius (ft)

Thrust = Gross weight (lb) during free flight in which there is not acceleration or velocity component in the vertical direction; tether load must be added in the case of tethered hover.

V_T = True airspeed (kt)

a = Speed of sound (ft/sec)

True airspeed (V_T) was calculated using calibrated airspeed (V_{cal}) and density ratio (σ) as follows:

$$V_T = V_{cal}/\sqrt{\sigma} \tag{5}$$

Shaft Horsepower Required

4. Engine output shaft torque was determined by the use of the engine torquemeter (para 2, app D). This torquemeter was calibrated in a test cell by the engine manufacturer (figs. 197 and 198, app G). The output from the engine torquemeter was recorded on the on-board data recording system. The output shp was determined from the engine output shaft torque and rotational speed by the following equation:

$$SHP = \frac{(2\pi)\;(N_p)\;(Q)}{33,000} \tag{6}$$

Where:

Np = Engine output shaft rotational speed (rpm)

Q = Engine output shaft torque (ft-lb)

Tail Rotor Performance

5. During hover performance tests, tail rotor performance parameters were also recorded. Terms in equations 1 and 2, which apply to the main rotor, were replaced by tail rotor parameters for nondimensionalized tail rotor performance. The terms redefined are as follows:

SHP = Tail rotor shaft horsepower (SHP_{TR})

A = Tail rotor disc area (ft^2)

Ω = Tail rotor angular velocity (radian/sec)

R = Tail rotor radius (ft)

Thrust = Tail rotor thrust (lb)

Tail rotor shp was determined from the following equation:

$$SHP_{TR} = \frac{(2\pi)(N_R)(4.8823)(Q_{TR})}{33,000} \qquad (7)$$

Where:

Q_{TR} = Tail rotor torque (ft-lb)

6. The tail rotor thrust for hover was determined by making two assumptions. These assumptions were necessary since sufficient information was not available and tail rotor thrust could not be measured directly during the evaluation. The first assumption was that all directional moments to maintain stabilized hover would be generated by the antitorque tail rotor. This assumption neglected any possible restoring moments that could be derived from rotor downwash and recirculating airflow over the fuselage, tail boom section, and empennage. The second assumption was that the temperature of the airflow passing through the tail rotor was not significantly influenced by engine exhaust gases. Tail rotor thrust was determined from the following equation:

$$\text{Thrust} = \frac{Q_{MR}}{l_t} \qquad (8)$$

Where:

QMR = Main rotor shaft torque (ft-lb)

l_t = Perpendicular distance between center lines of main and tail rotor shafts = 28.49 feet.

Hover

7. Hover performance was obtained both IGE and OGE by the tethered hover technique. Additional free flight hover data were accumulated to verify the tethered hover data. All hover tests were conducted in winds of less than 3 knots. Tethered hover consisted of restraining the helicopter to the ground by a cable in series with a load cell. An increase in cable tension, measured by the load cell, had the same effect on hover performance as increasing gross weight. Free flight hover tests consisted of stabilizing the helicopter at a desired height with reference to the radar altimeter. Atmospheric pressure, temperature, and wind velocity were recorded from a ground weather station. All hover data were reduced to nondimensional parameters of C_P and C_T (equations 1 and 2, respectively), and grouped according to wheel height. A line was faired through each set of data. Summary hovering performance was then calculated from these nondimensional plots and the power available at selected ambient conditions and power settings.

Vertical Climb

8. The vertical climb technique used was to stabilized in a 100-foot OGE hover based on the radar altimeter and then to increase engine power by a predetermined increment of engine torque. Various increments of engine torque up to the engine topping limits were used. An Elliott low-airspeed system was used to provide cues of longitudinal and lateral translation. The Elliott low-airspeed system was accurate to within 2 knots in horizontal airspeed. Each vertical climb was flown at a predetermined C_T with rotor speed held constant at 100 percent (289 rpm). Ballast was added as fuel burned off or temperature varied. Tests were conducted in ambient wind conditions of 3 knots or less.

9. Climb rates were measured after the aircraft was stabilized in unaccelerated vertical climbing flight primarily by means of a radar altimeter and the vertical speed transducer in the Elliott air data computer. Vertical climb performance is presented in terms of the generalized and referred parameters shown below.

a. Vertical velocity ratio (VVR):

$$VVR = V_v/\Omega R \sqrt{CT/2} \qquad (9)$$

b. Generalized power variation from hover:

$$\Delta CP_{gen} = (CP_c - CP_h)/.707\ CT^{3/2} \tag{10}$$

c. Referred vertical rate of climb (R/C_R):

$$R/C_R = R/C/\sqrt{\theta} \tag{11}$$

d. Referred shaft horsepower (SHP_R)

$$SHP_R = SHP/\delta\sqrt{\theta} \tag{12}$$

e. Referred gross weight (GW_R):

$$GW_R = GW/\delta \tag{13}$$

Forward Flight Climb

10. The forward flight climb tests were conducted at a constant rotor speed and a predetermined power and airspeed schedule. The power schedule represented single-engine IRP available at 35°C based on the engine model specification corrected for engine installation losses. The climb airspeed schedule was the airspeed corresponding to $V_{min\ pwr}$. Two continuous single-engine climbs to service ceiling were accomplished at reciprocal headings to average the effect of wind shear on climb performance. The data were corrected from test-day conditions to 35°C at all altitudes; from test-day gross weight to a constant design gross weight of 14,242 pounds; and for any deviations from the climb airspeed and power schedules. Test rate of climb was determined from pressure altitude variation with time and corrected for altimeter error caused by nonstandard temperature using the following equation:

$$R/C_T = \left(\frac{dH_p}{dt}\right)\left(\frac{T_t}{T_s}\right) \tag{14}$$

Where:

R/C_T = Tapeline rate of climb (ft/min)

$\frac{dH_p}{dt}$ = Slope of pressure altitude versus time curve at a given pressure altitude (ft/min)

T_t = Test ambient air temperature at the pressure altitude at which slope is taken (°K)

T_s = Standard ambient air temperature at the pressure altitude at which slope is taken (°K)

Tapeline rate of climb and test shp were referenced to test-day density altitude. All corrections to rate of climb were applied with reference to density altitude. Power corrections were made by the following equation:

$$\Delta R/C_P = \frac{(K_p)\ (\Delta SHP)\ (33{,}000)}{GW_t} \tag{15}$$

Where:

K_p = Power correction factor

$\Delta SHP = SHP_s - SHP_t$

GW_t = Test gross weight (lb)

Where:

SHP_s = SHP at 35°C ambient temperature available from model specification engine installed

SHP_t = Test shp measured

Weight corrections were made by the following equation:

$$\Delta R/C_w = (K_w)\ (SHP_s)\ (33{,}000)\left[\frac{GW_t - GW_s}{(GW_s)(GW_t)}\right] \tag{16}$$

Where:

K_w = Weight correction factor

GW_s = Standard gross weight (lb)

11. Two additional series of climbs were conducted to achieve the power and weight correction factors. These factors were used to correct the continuous climb data from test to desired conditions, as mentioned in the previous paragraph. The K_P climbs were conducted at a constant aim gross weight from pressure altitudes of 2300 feet to 4000 feet at various power settings. The K_W climbs were flown partial power at various gross weights from 3000 to 5000 feet pressure altitude. These climbs were corrected for deviations from the aim power schedule. Corrected test results of rate of climb versus shp and gross weight are presented in figures 11 and 12, appendix G. A constant value of 0.80 was determined for K_P, while K_W was found to vary as a function of gross weight throughout the altitude range tested. Power and weight factors were determined from the following equations:

$$K_P = \left(\frac{\Delta R/C}{\Delta SHP}\right)\frac{GW_t}{33{,}000} \tag{17}$$

$$K_W = \frac{\Delta R/C}{\Delta GW} \times \frac{GW^2}{(SHP)\ (33{,}000)} \tag{18}$$

Where:

$\Delta R/C/\Delta SHP$ = Slope of rate of climb versus shp curve

$\Delta R/C/\Delta GW$ = Slope of rate of climb versus gross weight curve

GW = Gross weight at which slope was read (lb)

SHP = Power at which sawtooth climbs were flown

12. Power corrections were applied for variations in airspeed from the climb airspeed schedule determined from the level flight performance data. Any deviations from $V_{min\ pwr}$ were corrected by equation 17.

Where:

ΔSHP = Difference in test shp measured and $V_{min\ pwr}$ at test conditions

13. A power correction was applied for the increased airframe drag due to instrumentation, again using equation 17.

Where:

ΔSHP = Difference in test shp measured and $V_{min\ pwr}$ without airframe-mounted instrumentation

14. The standard rate of climb at 35°C ambient temperature was determined by the summarized equation:

$$R/C_s = R/C_T + \Delta R/C_P + R/C_w + \Delta R/C_{A/S} + \Delta R/C_{INSTR} \quad (19)$$

Where:

$\Delta R/C_{A/S}$ = Rate of climb difference due to deviation from climb airspeed schedule (ft/min)

$\Delta R/C_{INSTR}$ = Rate of climb difference due to instrumentation drag

The corrected R/C_S, SHP_S, and the calculated time to climb, nautical air miles traveled, and V_T parameters with reference to density altitude, were plotted as a function of pressure altitude at 35°C conditions (fig. 9, app G).

Level Flight Performance

15. Level flight speed-power performance was determined by using equations 1, 2, and 3. Each speed power was flown at a constant C_T and rotor speed. To maintain gross weight ratio to air density ratio (W/σ) constant, altitude was increased as fuel was consumed. Test-day level flight power was corrected to standard-day conditions by assuming that the test-day dimensionless parameters, C_{P_t}, C_{T_t}, and μ are independent of atmospheric conditions. Consequently, the standard-day dimensionless parameters C_{P_S}, C_{T_S}, and μ_S are identical to C_{P_t}, C_{T_t}, and μ_t, respectively. From equation 1, the following relationship can be derived.

$$SHP_s = SHP_t \left(\frac{\rho_s}{\rho_t}\right) \quad (20)$$

Where:

t = Test day

s = Standard day

16. Curves defined by the power required as a function of airspeed were plotted as Cp versus μ for a constant value of C_T. These curves were then joined by lines of constant μ to form a carpet plot. The reduction of this carpet plot into a family of curves, C_T versus C_P, for a constant μ value allows determination of the power required as a function of airspeed for any value of C_T.

17. The specific range (NAMPP) data were derived from the test level flight power required and fuel flow. The NAMPP curves were obtained from the power and airspeed from the level flight carpet plot and fuel flow from the engine model specification, corrected for installation losses, for the particular conditions. The following equation was used for determination of NAMPP.

$$\text{NAMPP} = \frac{V_T}{W_f} \qquad (21)$$

Where:

V_T = True airspeed (kt)

W_f = Fuel flow (lb/hr)

The system specification endurance missions were determined based on 5 percent conservatism applied to the fuel flow.

18. The tip Mach number of the advancing blade during level flight was determined from equation 4.

19. Summary level flight performance plots for standard-day and US Army hot day presentations were corrected for the effects of instrumentation drag. An equivalent Δf_e of 0.90 ft^2, as provided by HHC through AVSCOM, was used to account for the additional power required for the instrumentation drag, and was subsequently subtracted from the power measured for level flight. A rotor efficiency of 100 percent was used for all equivalent flat plate area calculations.

Autorotational Descent Performance

20. Autorotational descent performance data were acquired at variations in stabilized airspeeds with constant rotor speed and variations in rotor speed with constant airspeed. The tapeline rates of descent were calculated by the following equation.

$$\text{R/D tapeline} = \left(\frac{dH_p}{dt}\right)\left(\frac{T_t}{T_s}\right) \qquad (22)$$

Where:

$\frac{dH_p}{dt}$ = Change in pressure altitude per given time (ft/sec)

T_t = Test ambient air temperature (°K)

T_s = Standard ambient air temperature (°K)

Vertical Displacement

21. Vertical displacement tests were conducted to quantify the agility of the helicopter. The tests were conducted by stabilizing in coordinated steady-heading level flight at 140 KTAS, then rapidly displacing the helicopter vertically a distance of 200 feet, as described in paragraphs 21 and 22 of the Results and Discussion section, while determining the horizontal distance along the entry flight path required to complete the maneuver. A Fairchild camera was utilized to determine vertical distance.

Lateral Acceleration

22. The lateral acceleration performance was obtained by determining the time, distance, and average acceleration required to go from an OGE vertical climb to a 35-KTAS sideward velocity. Space positioning data were obtained by flight path integration of the on-board inertial reference instrumentation and ROI's.

$$a = \frac{V}{32.174t} \tag{23}$$

Where:

a = Average acceleration (g)

v = Velocity (59.1 ft/sec) (35 KTAS)

t = Time to 35 KTAS (sec)

32.1740 = Conversion factor from ft/sec^2 to g

Vibration

23. Vibration data were analyzed using a Spectral Dynamics Model SD301B real time spectral analyzer. The analyzer converted the data from the time domain (acceleration as a function of time) to the frequency domain (acceleration as a function of frequency). The data were analyzed using two frequency ranges: zero to 100 Hz, and zero to 500 Hz. Frequency resolution was 0.2 Hz for the 100-Hz range and 1 Hz for the 500-Hz range. In order to minimize random variation in acceleration amplitude, the data were averaged over an 8-second time interval using a Spectral Dynamics Model 302B ensemble averager.

Engine Performance

24. Engine temperature and pressure inlet characteristics were obtained from an inlet ring consisting of eight rakes with five total pressure sensors and three total temperature sensors on each rake (40 total pressure sensors and 24 total

temperature sensors). Four static pressure ports were also located in the same plane. The probes were manifolded into one average reading for each parameter. The pressure probes were referenced to the test boom static source system to provide a differential pressure measurement. The temperature probes measured temperature directly.

25. Engine inlet temperature rise is based on the total temperature at the probe minus the boom total temperature corrected to ambient conditions, as follows:

$$\Delta T = T_{T1} - T_{SO} \tag{24}$$

Where:

T_{T1} = Engine inlet total temperature (°C)

T_{SO} = Ambient temperature (°C)

26. Pressure ratio determined at the engine inlet is based on the total pressure at the probe divided by the test boom static pressure corrected to ambient conditions, as follows:

$$\frac{P_{T1}}{P_{SO}} = \frac{\Delta P_1 + P_{SB}}{P_{SO}} \tag{25}$$

Where:

$\Delta P_1 + P_{SB} = P_{T1}$, engine inlet total pressure (lb/in.^{2})

Where:

$\Delta P_1 = P_{T1} - P_{SB}$, differential pressure (lb/in.^{2})

P_{SB} = Static pressure (boom system) (lb/in.^{2})

$P_{SO} = P_{SB}$ + position error

Engine Exhaust System Performance

27. Engine exhaust loss was determined by measuring the average static pressure at the exhaust station (P_{S7}), power turbine measured gas temperature ($T_{4.5}$), power turbine speed (N_P), and inlet pressure and temperatures (P_{T1} and T_{T1}). The procedures for reduction of the test data were provided by AVSCOM in

reference 11, appendix A. Figures 1 and 2 of reference 11 provided the referred velocity head at station 7 ($VH7/\delta 1$) and exhaust swirl angle (Γ) as a function of referred measured gas temperature ($T4.5/\theta 1^{.96}$) and referred power turbine speed ($Np/\sqrt{\theta 1}$). The exhaust duct pressure recovery coefficient (PS9D7Q) was computed by equation 26 and presented as a function of $\Gamma 7$ in the engine performance section. This curve was input to the YT700-GE-700 engine deck No. 73004 to determine power available.

$$PS9D7Q = \frac{(P_{SO} - P_{s7})/\delta_1}{VH7/\delta_1} + 0.4070 \sin^2\Gamma_7 \qquad (26)$$

Drive Train Losses

28. Main transmission and drive train power losses were determined by comparing the total engine shp to the total rotor horsepower, as follows.

$$\Delta_{HP} = ESHP - RHP \qquad (27)$$

Where:

ESHP = Total engine shp (both)

RHP = Main rotor horsepower plus tail rotor horsepower

Referred Engine Performance

29. The referred terms of the engine parameters were used to compare the test engines with the model specification engine. Data on shp, $T_{4.5}$, fuel flow, N_G, and compressor discharge static pressure were referred as follows.

a. Referred shp (SHP_R):

$$SHP_R = \frac{SHP}{(\delta_1)(\theta_1)^{0.50}} \ (SHP) \qquad (28)$$

b. Referred gas temperature ($GAST_R$)

$$GAST_R = \left[\frac{T_{4.5} + 273.15}{(\Theta_1)^{0.96}}\right] - 273.15 \ (°C) \qquad (29)$$

c. Referred fuel flow (WF_R):

$$WF_R = \frac{W_f}{(\delta_1)(\Theta_1)^{0.55}} \text{ (lb/hr)} \quad (30)$$

d. Referred gas generator speed (NG_R):

$$NG_R = \frac{N_G}{(\Theta_1)^{0.50}} \text{ (\%)} \quad (31)$$

e. Referred compressor discharge static pressure ($CDSP_R$):

$$CDSP_R = \frac{CDP + P_{SO}}{\delta_1} \text{(lb/in.}^2\text{)} \quad (32)$$

Where:

$$\delta_1 = \frac{\Delta P_1 + P_{SB}}{14.697}$$

$$\theta_1 = \frac{T_{T1} + 273.15}{288.15}$$

$T_{4.5}$ = Turbine inlet temperature (°C)

W_f = Engine fuel flow (lb/hr)

N_G = Gas generator speed referenced to 44,700 rpm (100 percent)

$CDP + P_{SO}$ = Compressor discharge static pressure (lb/in.^{2})

30. The specification shp available and specification fuel flow were obtained from the General Electric engine specification. Inlet and exhaust losses were determined from figures 164 through 173, appendix G, and input in YT700-GE-700 engine deck No. 73004 to obtain installed data.

AIRSPEED CALIBRATION

31. The boom and ship's standard pitot-static system were calibrated by using the pace aircraft method to determine airspeed position error. Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}) and position error (ΔV_{pc}).

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \tag{33}$$

32. True airspeed (V_T) was calculated from the calibrated airspeed and density ratio.

$$V_T = \frac{V_{cal}}{\sqrt{\sigma}} \tag{34}$$

Where:

σ = Density ratio ($\frac{\rho}{\rho_0}$ where ρ is the test ambient density).

WEIGHT AND BALANCE

33. Prior to testing, the aircraft gross weight and longitudinal and lateral cg were determined utilizing accurate calibrated scales capable of allowing the aircraft landing gear loads to be determined independently.

34. The longitudinal cg was calculated by a summation of moments about a reference datum line (longitudinal station 0.0). The aircraft was weighed empty in the clean configuration, which included instrumentation minus all munitions and fuel. The basic aircraft weight was 11,600 pounds with a longitudinal cg location of 213.45 inches for aircraft SN 74-22248 and 11,754 pounds and 218.63 inches for aircraft SN 74-22249.

APPENDIX G. TEST DATA

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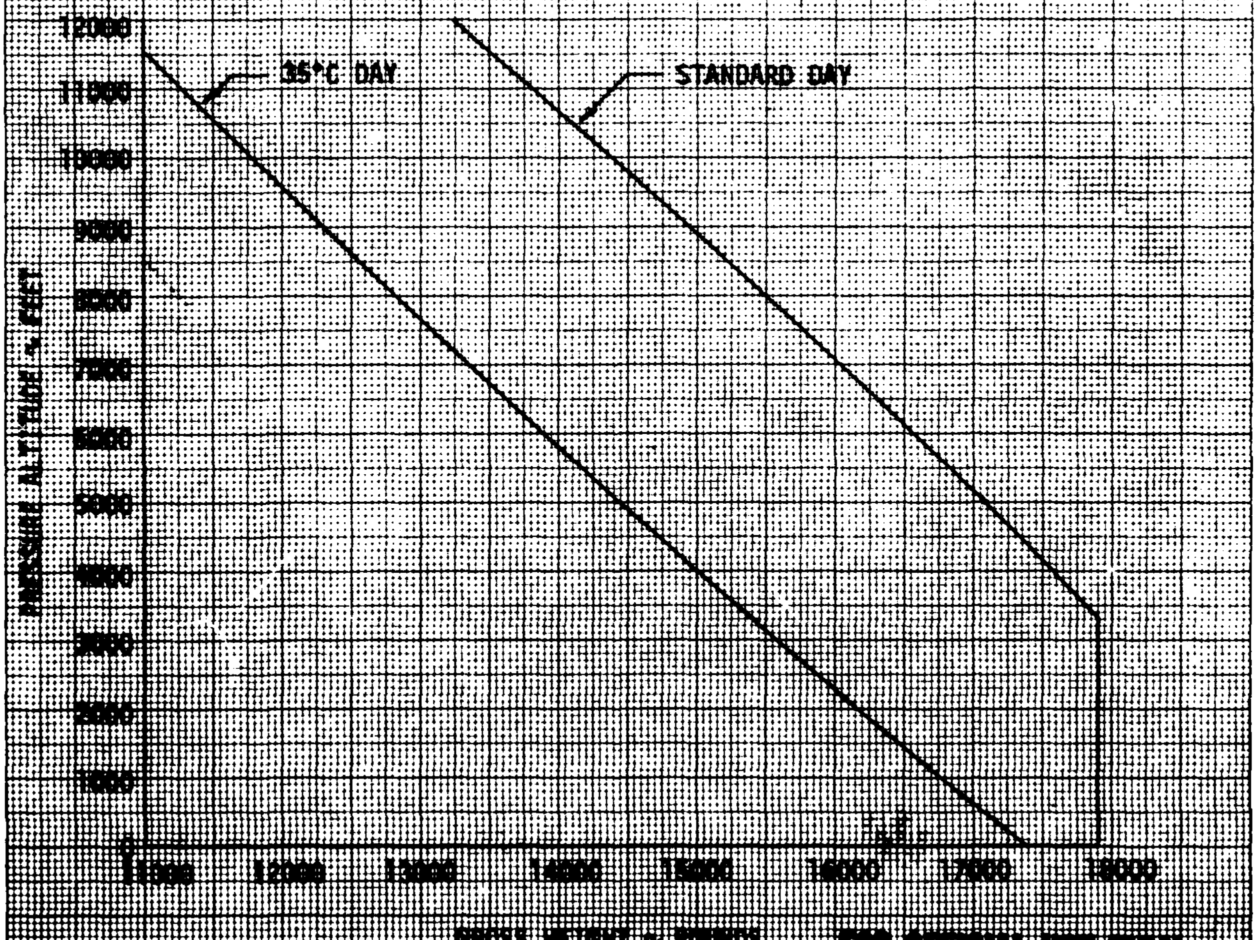
FIGURE 1
OGE HOVER CEILING
YAH-64 USA S/N 74-22249
YT-700-A-700 ENGINES

NOTE: 1. ROTOR SPEED 289 RPM
2. MAXIMUM GROSS WEIGHT LIMIT OF 17900 POUNDS FROM SAFETY-OF-FLIGHT RELEASE (Reference 4, appendix A)
3. CLEAR CONFIGURATION
4. INTERMEDIATE RATED POWER OBTAINED FROM FIGURE



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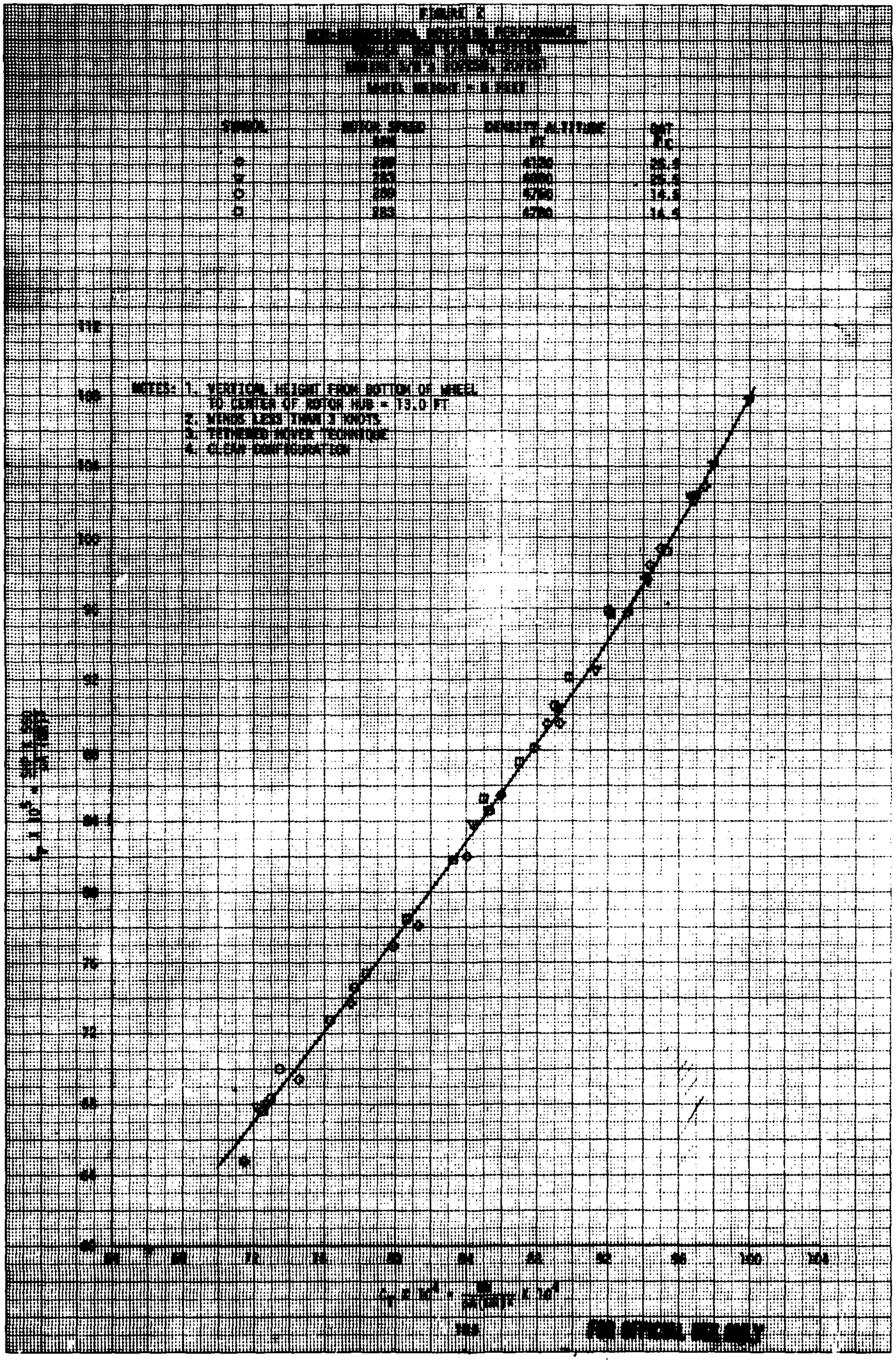

NOTES: 1. VERTICAL HEIGHT FROM BOTTOM OF WHEEL
TO CENTER OF ROTOR HUB = 13.0 FT
2. WINDS LESS THAN 3 KNOTS
3. TETHERED HOVER TECHNIQUE
4. CLEAN CONFIGURATION
FOR OFFICIAL USE ONLY

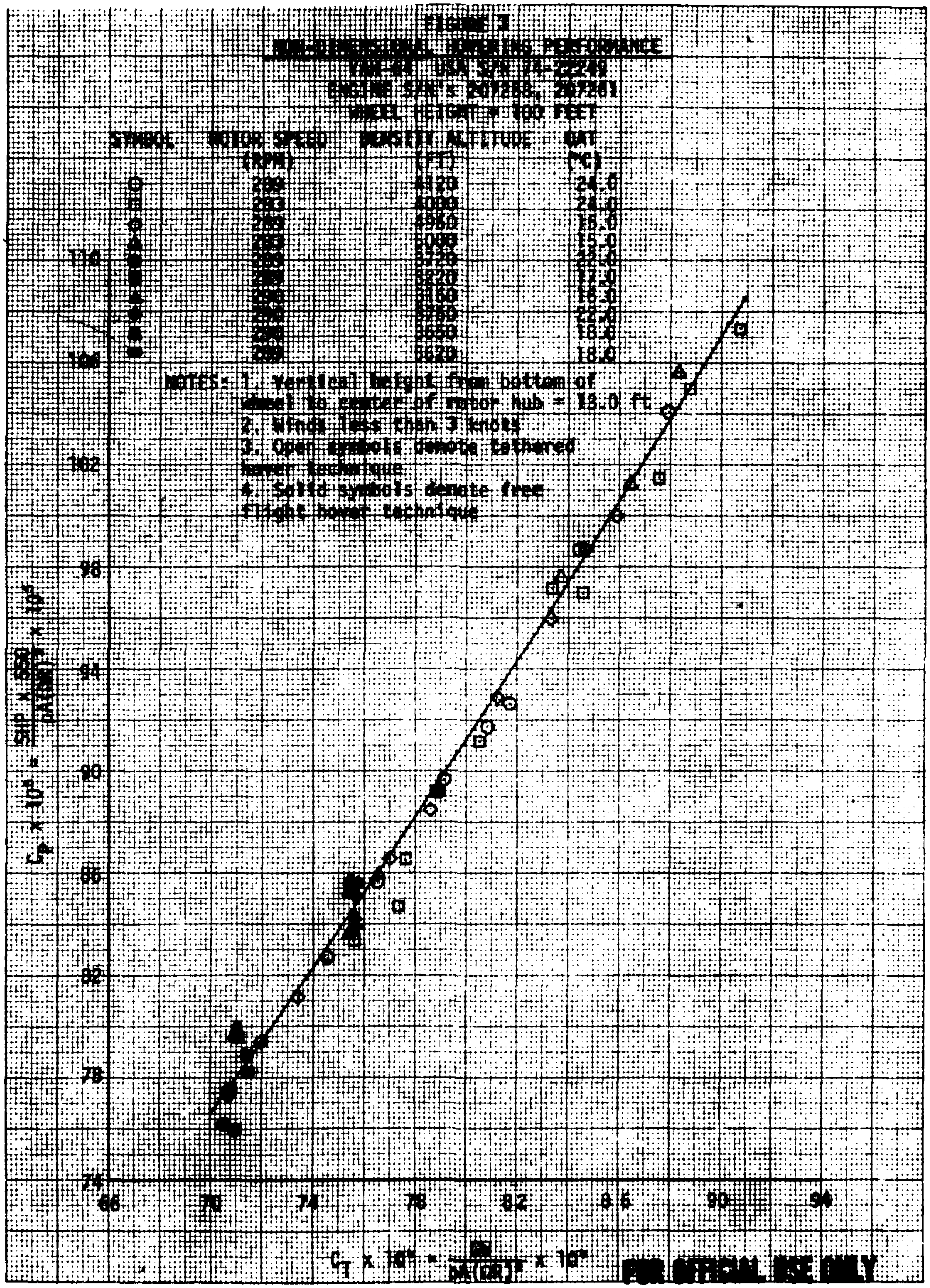


FIGURE 3
NON-DIMENSIONAL HOVERING PERFORMANCE
YAH-64 USA S/N 74-22249
ENGINE S/N's 207258, 207261
WHEEL HEIGHT = 100 FEET
SYMBOL ROTOR SPEED (RPM) DENSITY ALTITUDE (FT) OAT (°C)
NOTES: 1. Vertical height from bottom of wheel to center of rotor hub = 13.0 ft
2. Winds less than 3 knots
3. Open symbols denote tethered hover technique
4. Solid symbols denote free flight hover technique
110
106
102
98
94
90
86
82
78
74
66
70
74
78
82
86
90
94
$C_P \times 10^4 = \frac{SHP \times 550}{\rho A(\Omega R)^3} \times 10^4$
$C_T \times 10^4 = \frac{GW}{\rho A(\Omega R)^2} \times 10^4$
FOR OFFICIAL USE ONLY

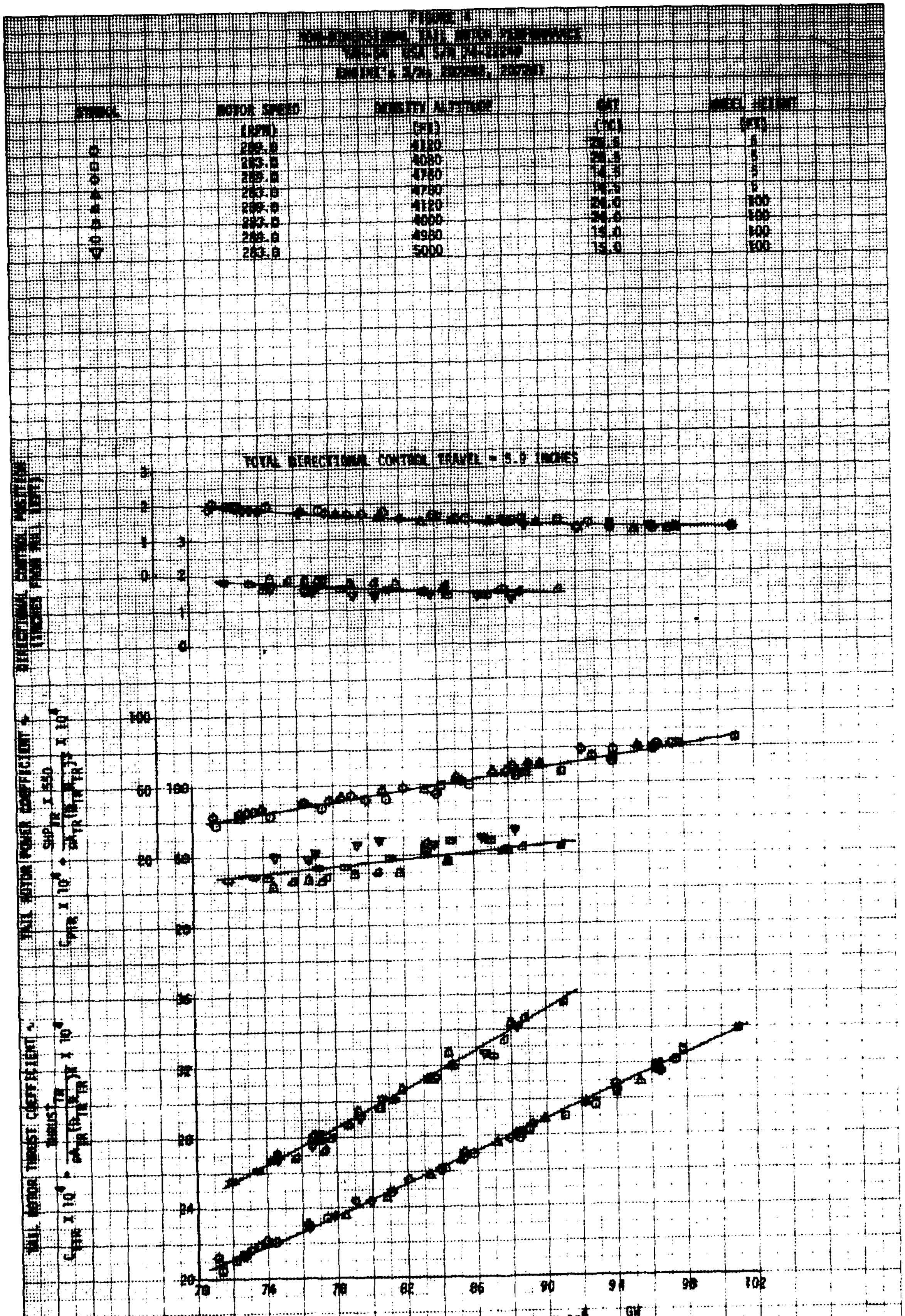
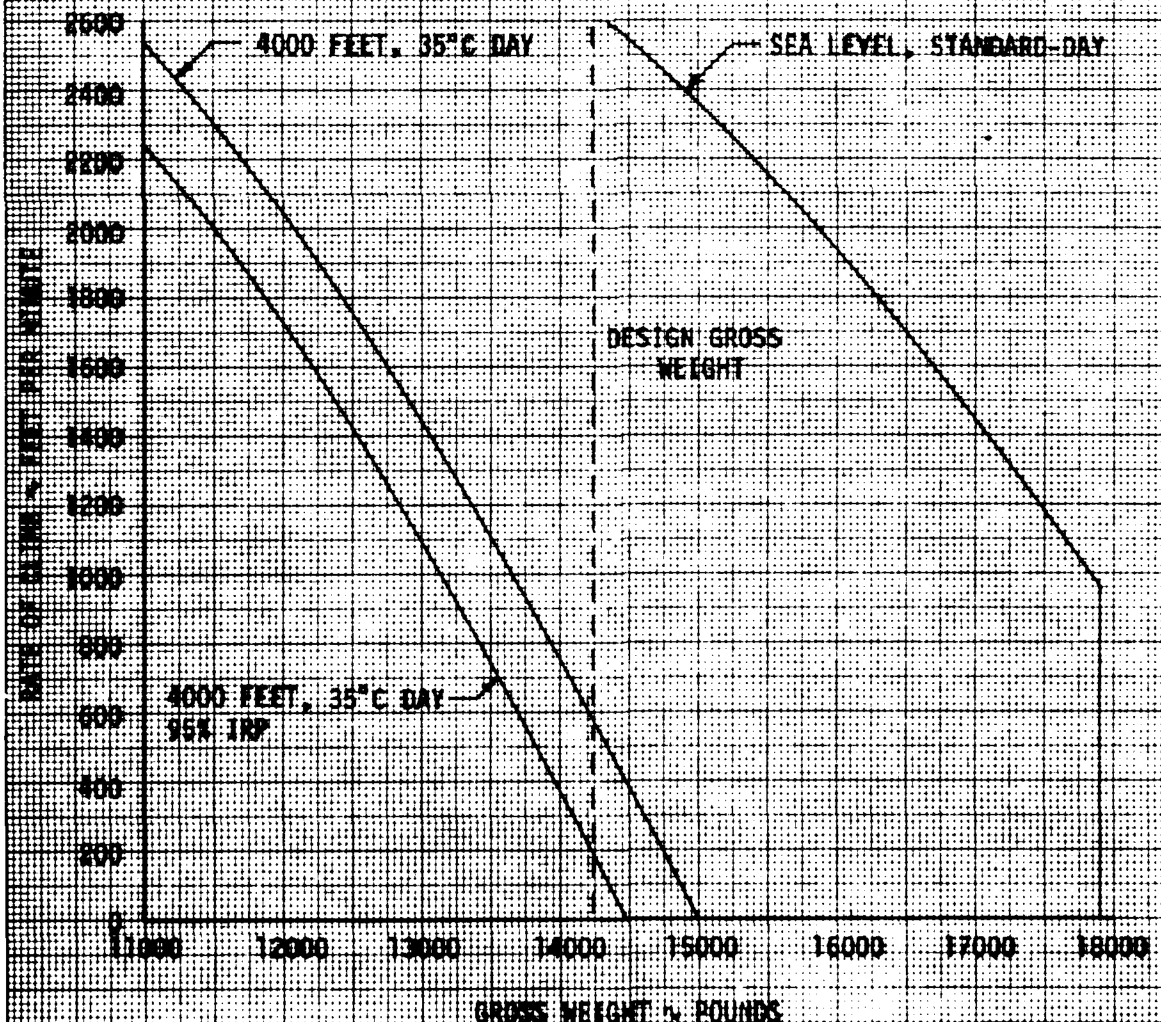

SYMBOL
ROTOR SPEED
(RPM)
DENSITY ALTITUDE
(FT)
4120
4080
4780
4780
4120
4000
4980
5000
OAT
WHEEL HEIGHT
(FT)
100
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.9 INCHES
MAIN ROTOR THRUST COEFFICIENT ~ C_T X 10^4
70
74
78
82
86
90
94
98
102

FIGURE 5
VERTICAL CLIMB PERFORMANCE SUMMARY
YAH-64 USA S/N 74-22249

NOTE: 1. MAXIMUM RATE OF CLIMB AT INTERMEDIATE RATED POWER (IRP) UNLESS OTHERWISE NOTED.
2. ROTOR SPEED = 289 RPM
3. SPECIFICATION POWER BASED ON T700-GE-700 PID SPECIFICATION AMC-CP-2222-02000, 2 Feb 73.
4. RATE OF CLIMB CALCULATED FROM FIGURE 6, APPENDIX C.
5. MAXIMUM GROSS WEIGHT LIMIT OF 17900 POUNDS FROM SAFETY-OF-FLIGHT RELEASE (Ref 4, app A)



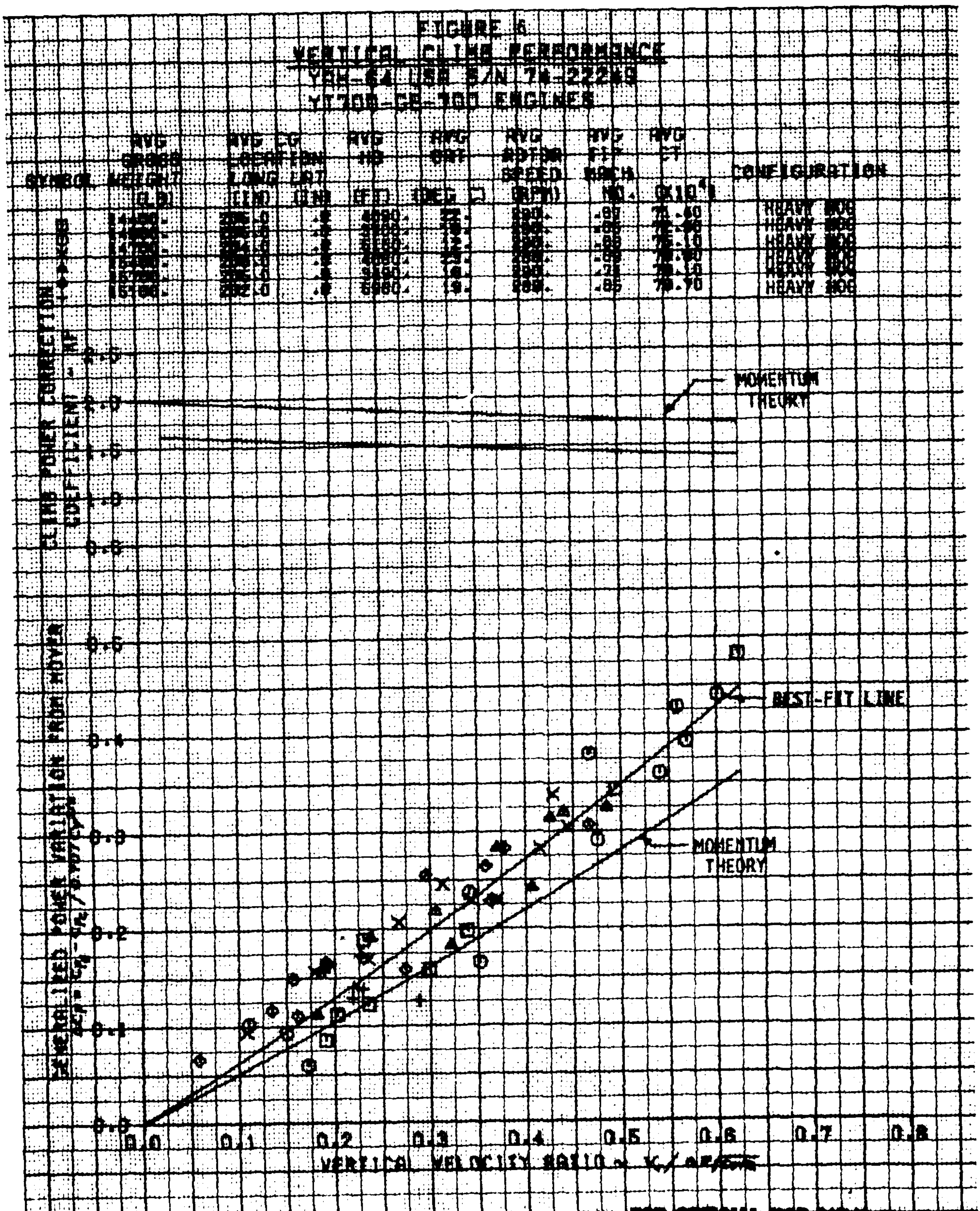


FIGURE 6
VERTICAL CLIMB PERFORMANCE
YT700-GE-700 ENGINES
CONFIGURATION
MOMENTUM THEORY
BEST-FIT LINE
MOMENTUM THEORY

FIGURE 7
VERTICAL CLIMB PERFORMANCE
YAH-64 USA S/N 74-22249
YT700-GE-700 ENGINES

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG HD (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG TIP MACH NO.	AVG CT $(X10^4)$	CONFIGURATION
[illegible]	14400.	205.8	.0	4090.	22.	290.	.87	71.45	HEAVY HOG
[illegible]	14900.	205.8	.0	3700.	18.	290.	.86	72.91	HEAVY HOG
[illegible]	14700.	204.6	.0	5150.	12.	290.	.88	75.13	HEAVY HOG

NOTE: DATA FAIRINGS BASED ON BEST-FIT LINE ON FIGURE 6

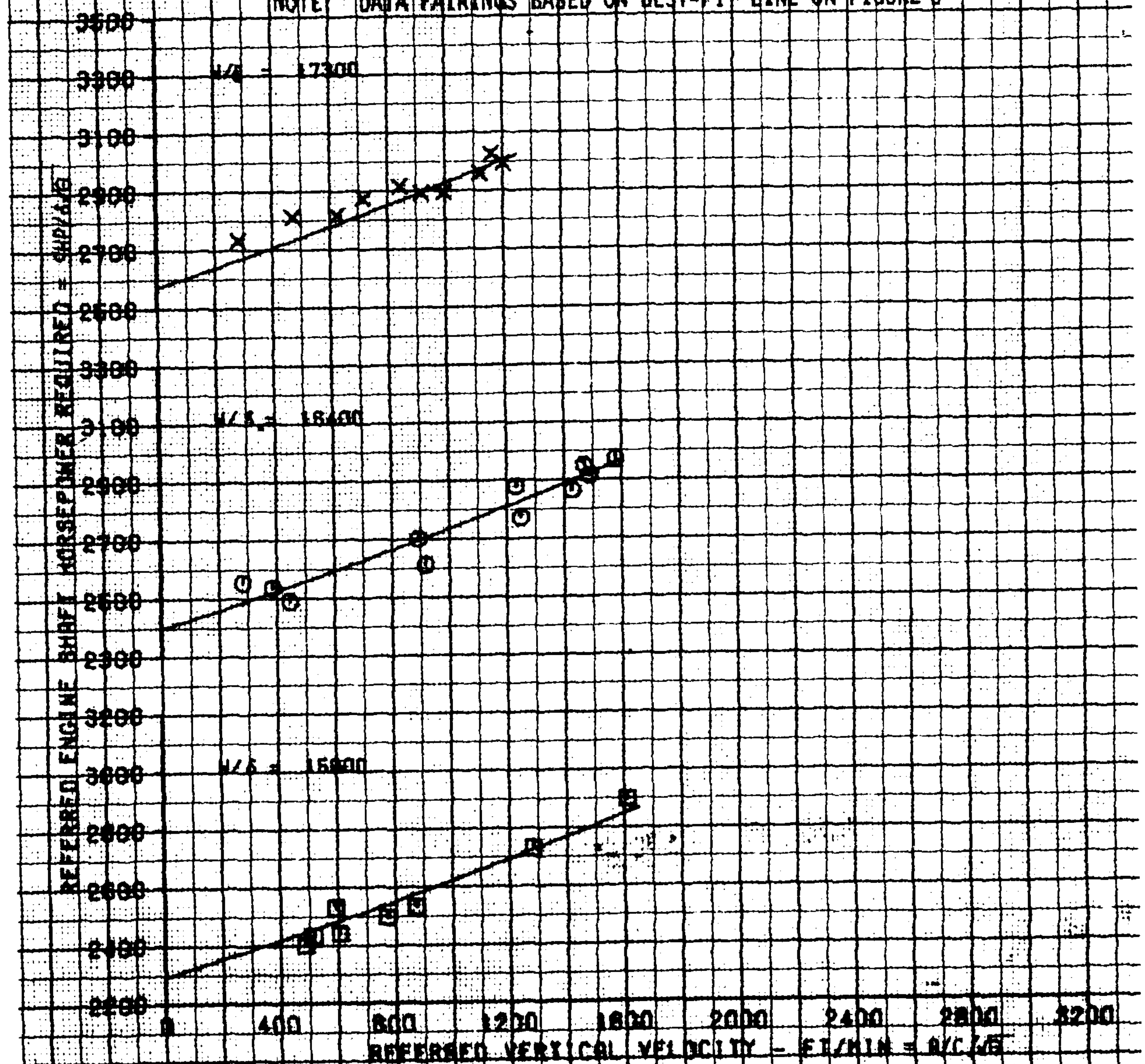


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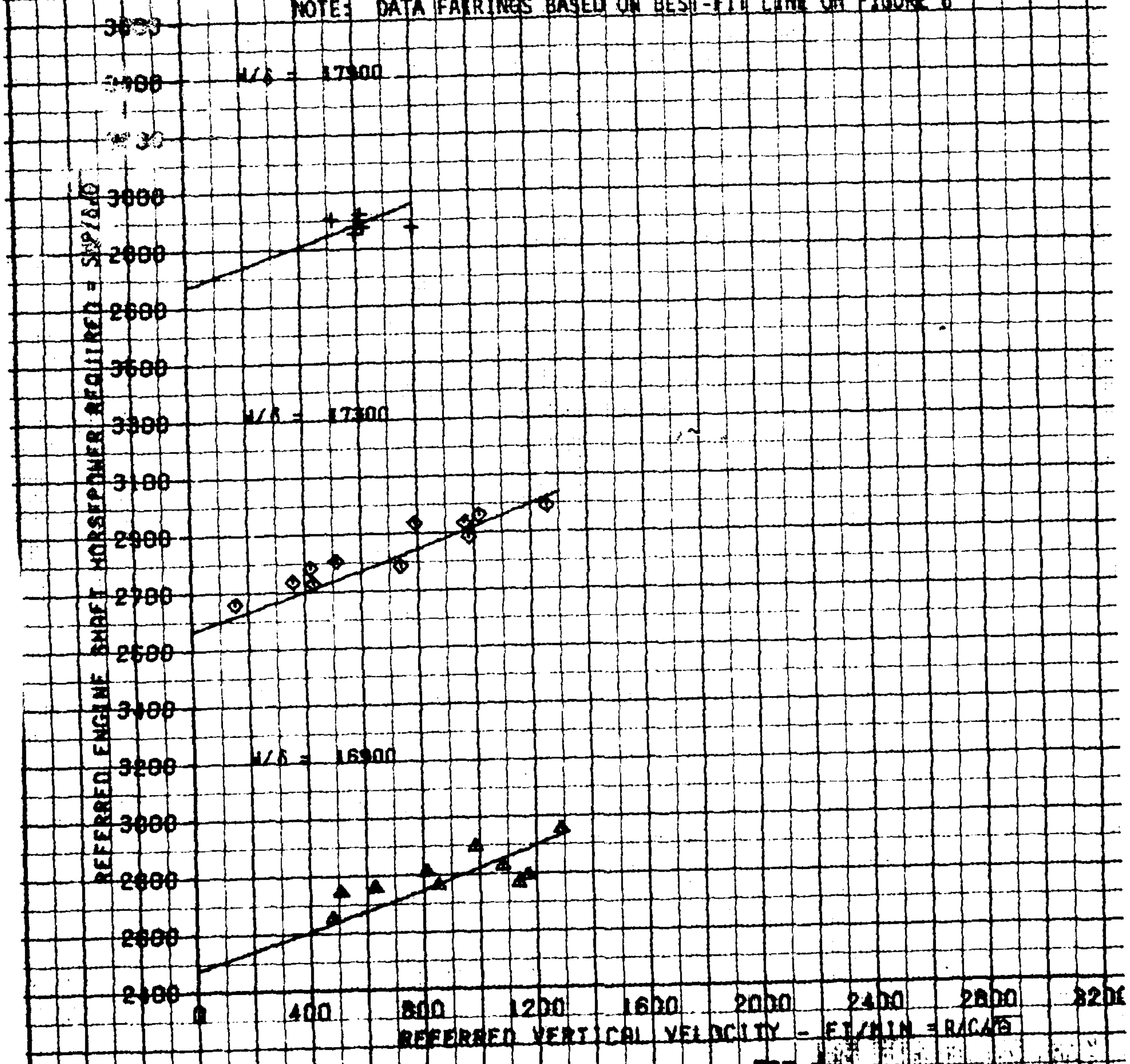
FIGURE 8
VERTICAL CLIMB PERFORMANCE
YAH-64 USA S/N 74-22249
YT700-GE-700 ENGINES

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG HD (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG TIP MACH NO.	AVG CT (X10⁴)	CONFIGURATION
▲	15400.	206.0	.0	4080.	23.	288.	.69	78.78	HEAVY HOG
◇	15700.	205.8	.0	3990.	18.	290.	.71	78.14	HEAVY HOG
+	15100.	202.0	.0	5980.	19.	289.	.65	78.74	HEAVY HOG

NOTE: DATA FAIRINGS BASED ON BEST-FIT LINE ON FIGURE 6



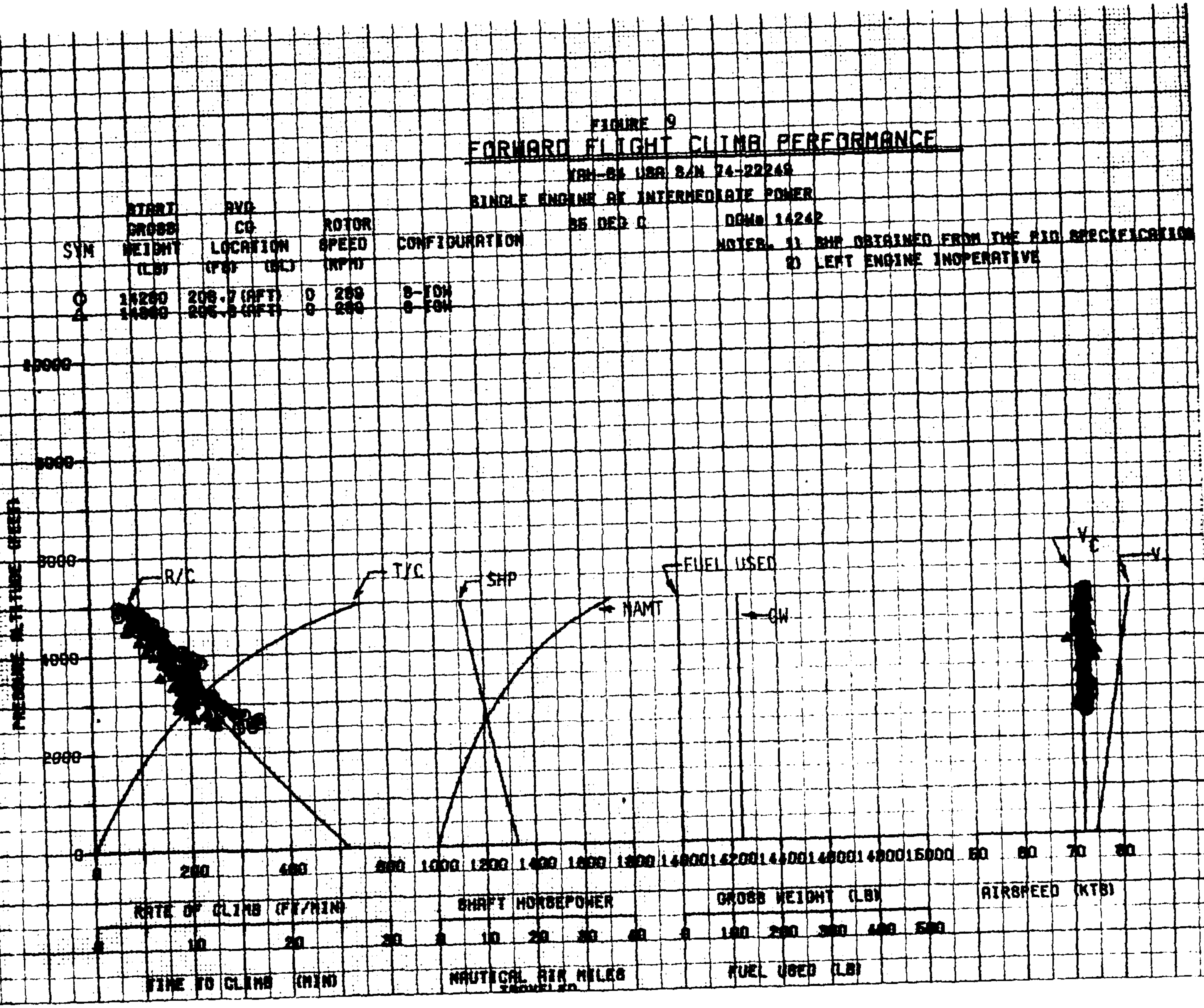


FIGURE 9
FORWARD FLIGHT CLIMB PERFORMANCE
YAH-64 USA S/N 74-22249
SINGLE ENGINE AT INTERMEDIATE POWER
START GROSS WEIGHT (LB)
AVG CG LOCATION (FS) (BL)
ROTOR SPEED (RPM)
CONFIGURATION
SYM
14200 206.7(AFT) 0 289 8-TOW
14000 206.5(AFT) 0 289 8-TOW
35 DEG C
DA= 14242
NOTES: 1) SHP OBTAINED FROM THE AID SPECIFICATION
2) LEFT ENGINE INOPERATIVE
PRESSURE ALTITUDE (FEET)
R/C
T/C
SHP
NAMT
FUEL USED
GW
V_C
V
RATE OF CLIMB (FT/MIN)
TIME TO CLIMB (MIN)
SHAFT HORSEPOWER
NAUTICAL AIR MILES TRAVELED
GROSS WEIGHT (LB)
FUEL USED (LB)
AIRSPEED (KTS)

FIGURE 10
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF HORSEPOWER
YAH-64 S/N 74-22249

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT	TRIM CALIBRATED AIRSPEED (KT)
14550	205.6 (AFT)	6020	17.5	289.5	0.00764	73

NOTES: 1. $K_p = \frac{\Delta R/C}{\Delta SHP} \times \frac{GW}{33000}$

2. $K_p = 0.800$

3. 8-TOW CONFIGURATION

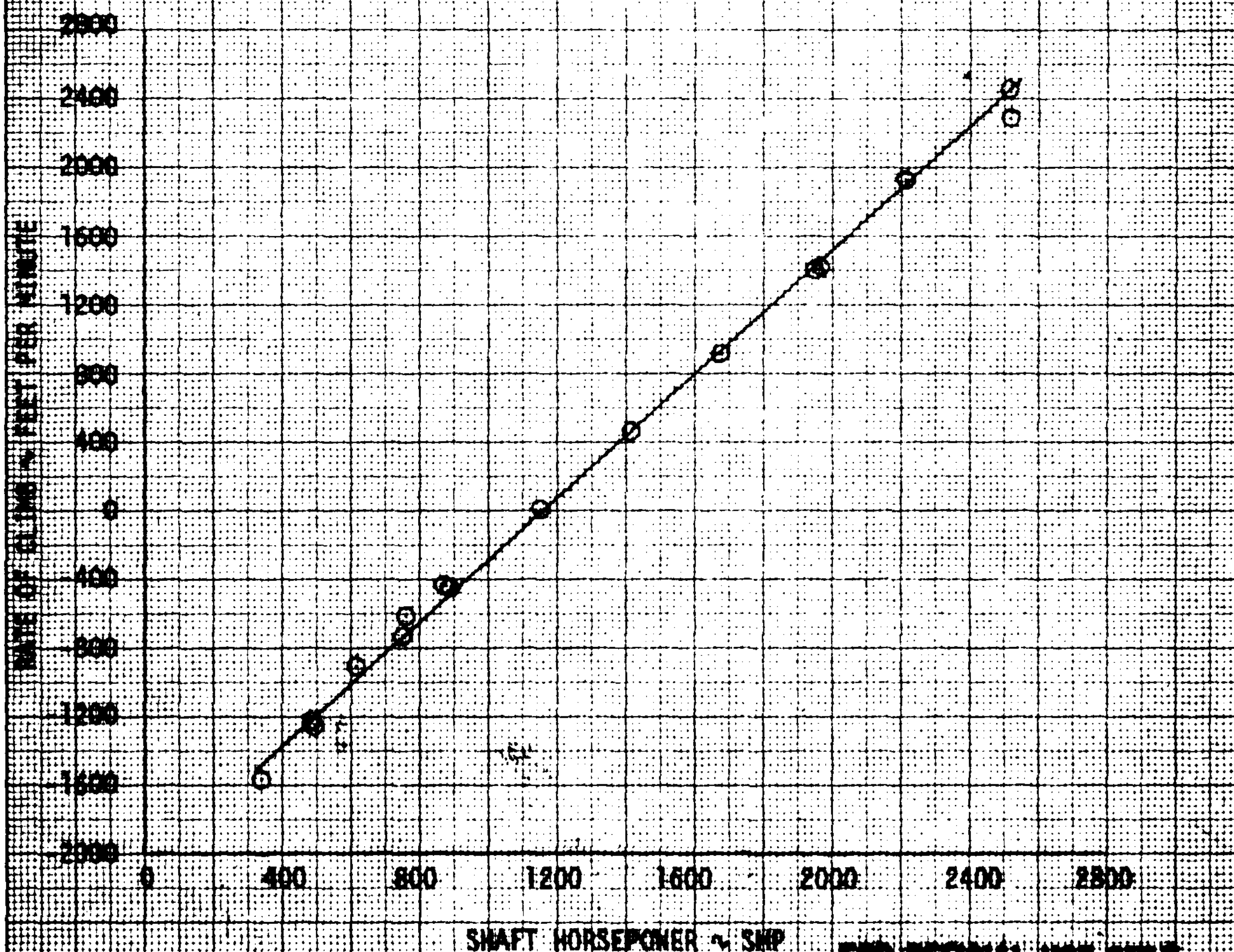


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FIGURE 11
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF GROSS WEIGHT
YAH-64 USA S/N 74-22249

AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SHAFT HORSEPOWER (SHP)
200.8 (AFT)	6200	18.0	289	73	1675

NOTE: 8-TOW CONFIGURATION

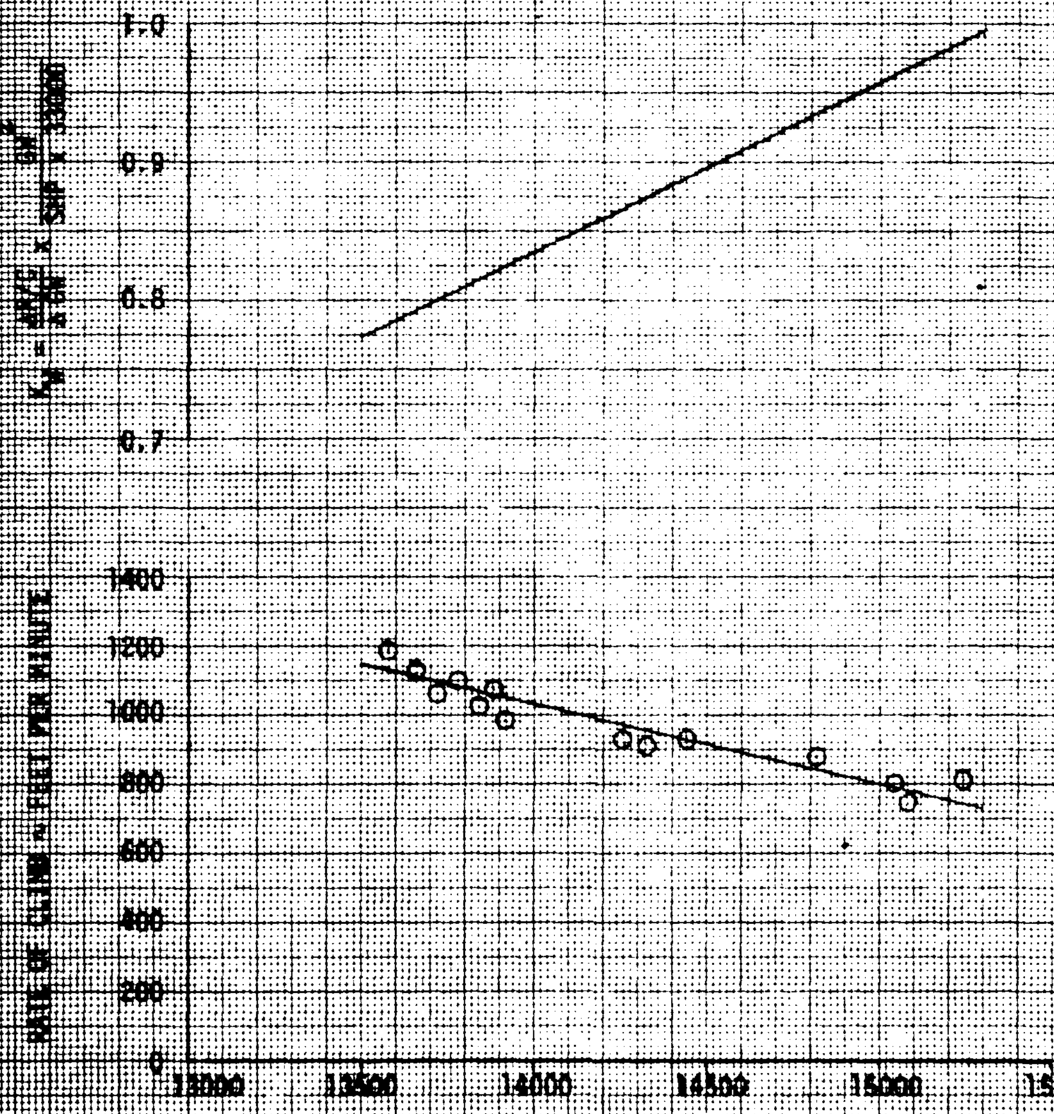

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FIGURE 12

NONDIMENSIONAL LEVEL-FLIGHT PERFORMANCE

YAH-64 USA S/N 74-22249

NOTES: 1. 8-TOW CONFIGURATION
2. FORWARD LONGITUDINAL CG
3. MID LATERAL CG
4. AVERAGE REFERRED ROTOR SPEED = 286.5 RPM $(N_R/\sqrt{\theta})$
5. CURVES DERIVED FROM FIGURES THRU .
6. ENGINE S/N's 207258 and 207261

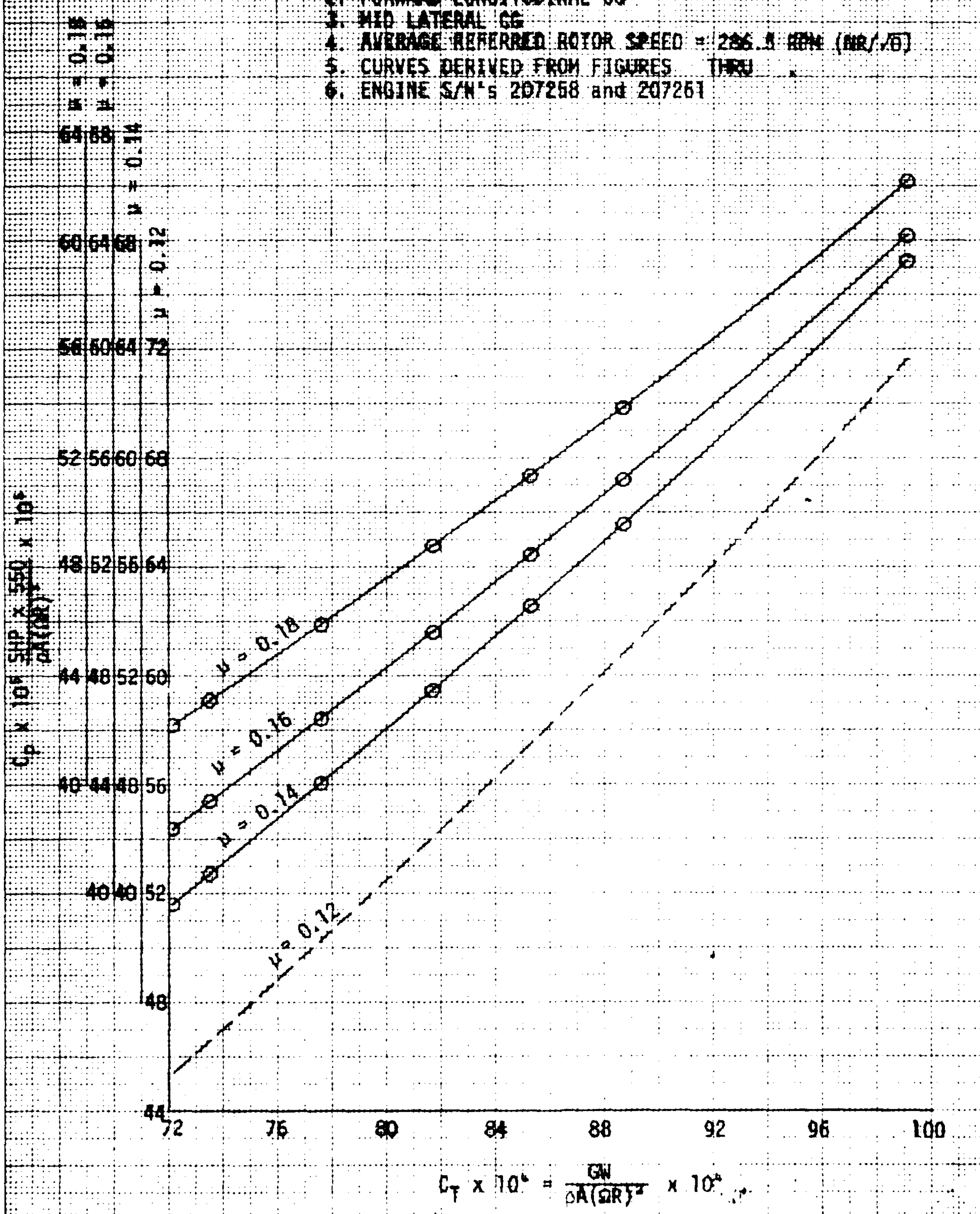

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FIGURE 13
NONDIMENSIONAL LEVEL-FLIGHT PERFORMANCE
YAH-64 USA S/N 74-22249

NOTES: 1. 8-TON CONFIGURATION
2. FORWARD LONGITUDINAL CG
3. MID LATERAL CG
4. AVERAGE REFERRED ROTOR SPEED = 286.5 RPM (NR/$\sqrt{\theta}$)
5. CURVES DERIVED FROM FIGURES THRU .
6. ENGINE S/N's 207258 and 207261

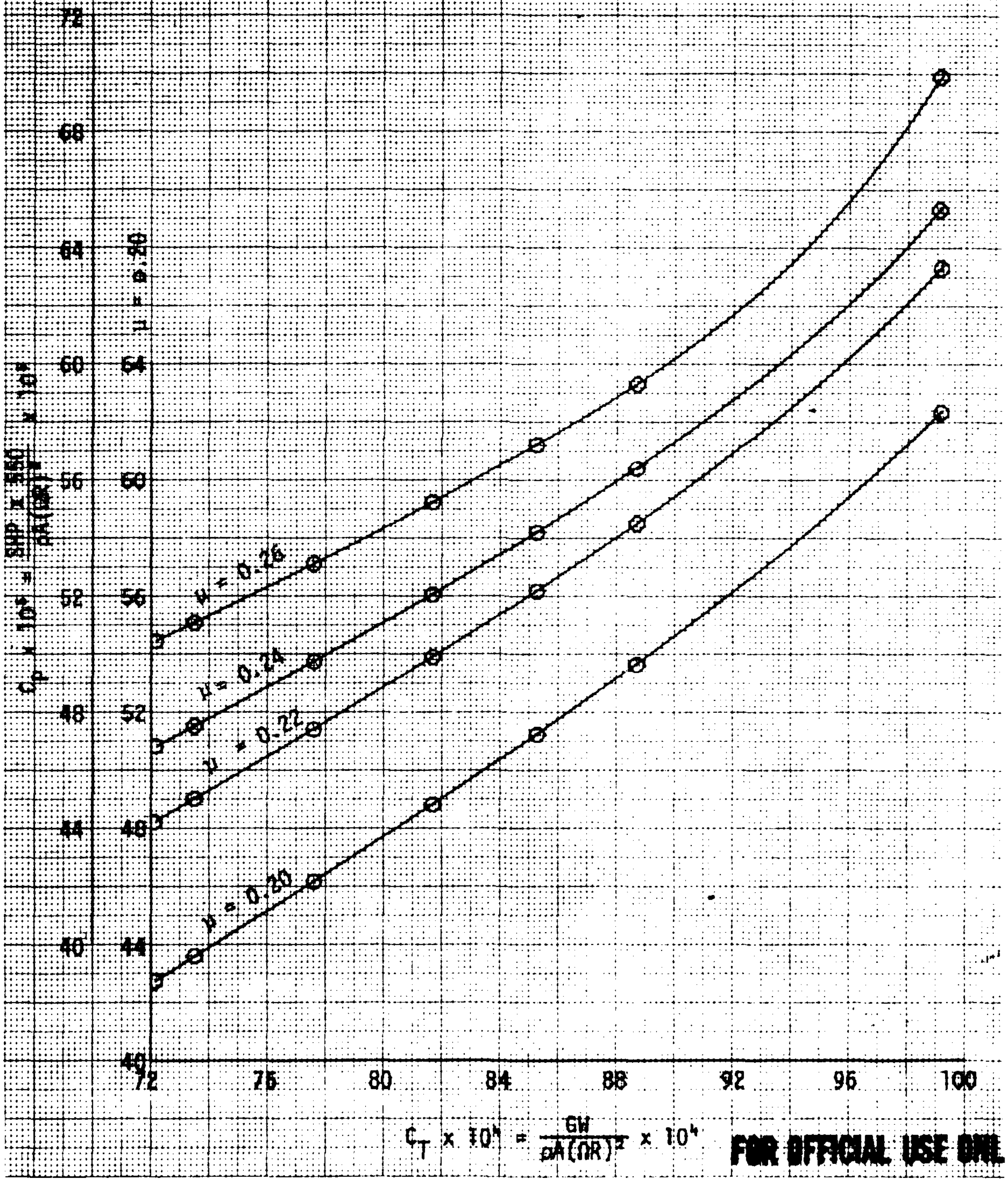

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FIGURE 14
NONDIMENSIONAL LEVEL-FLIGHT PERFORMANCE
YAH-64 USA S/N 74-22249

NOTES: 1. H-TOW CONFIGURATION
2. FORWARD LONGITUDINAL CG
3. MID LATERAL CG
4. [redacted] ROTOR SPEED = [redacted] (NR/√θ)
5. CURVES DERIVED FROM FIGURES THRU
6. ENGINE S/N's 207250 and 207261

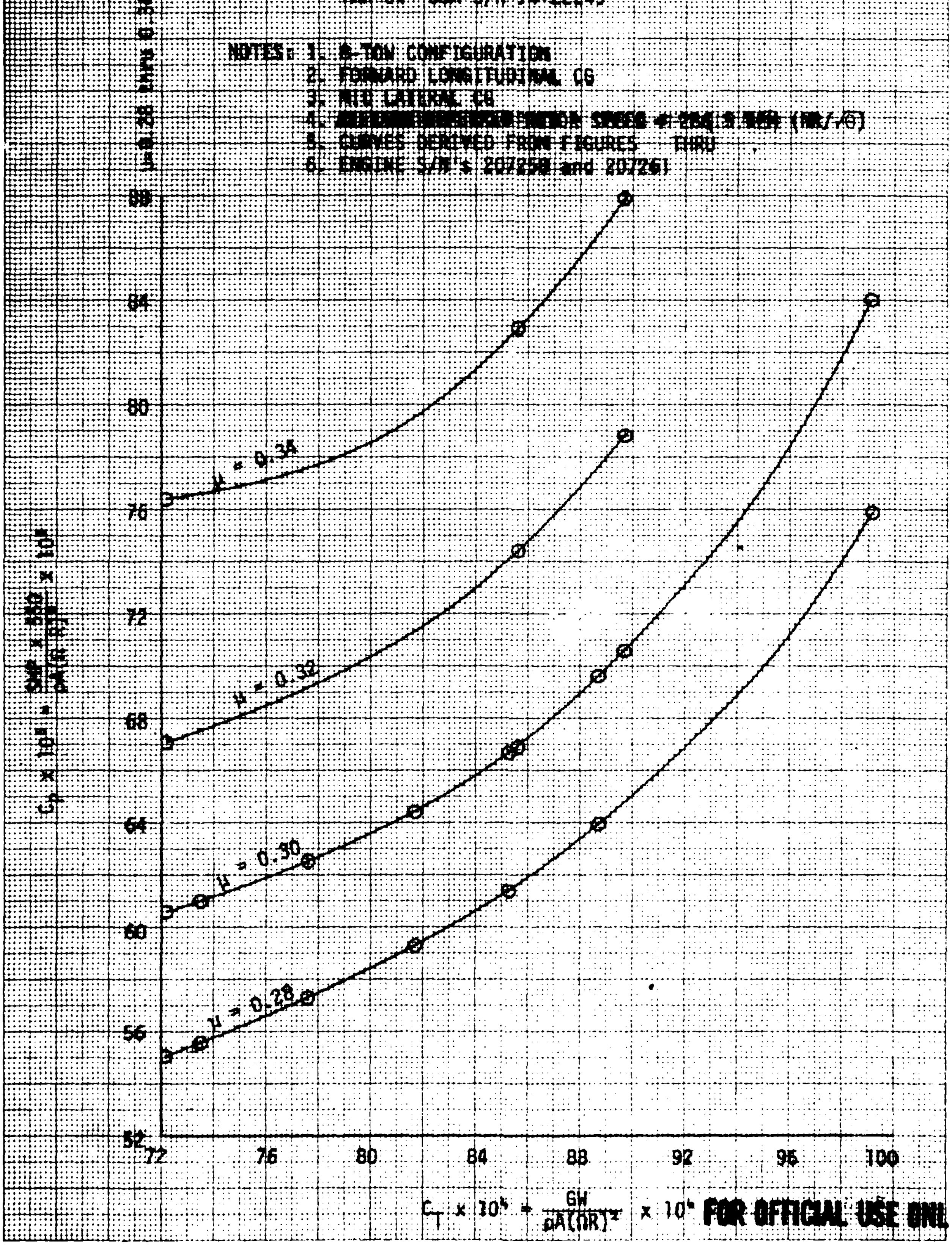


FIGURE 15
MAXIMUM SINGLE ENGINE LEVEL FLIGHT AIRSPEED
YAH-64 USA S/N 74-22249

NOTE:
1. 4000 FEET PRESSURE ALTITUDE, 35°C HOT DAY
2. ROTOR SPEED = 289 RPM
3. INTERMEDIATE RATED POWER (FIGURE)
4. 8-TOW CONFIGURATION
5. FORWARD CENTER-OF-GRAVITY
6. DATA DERIVED FROM FIGURES THRU

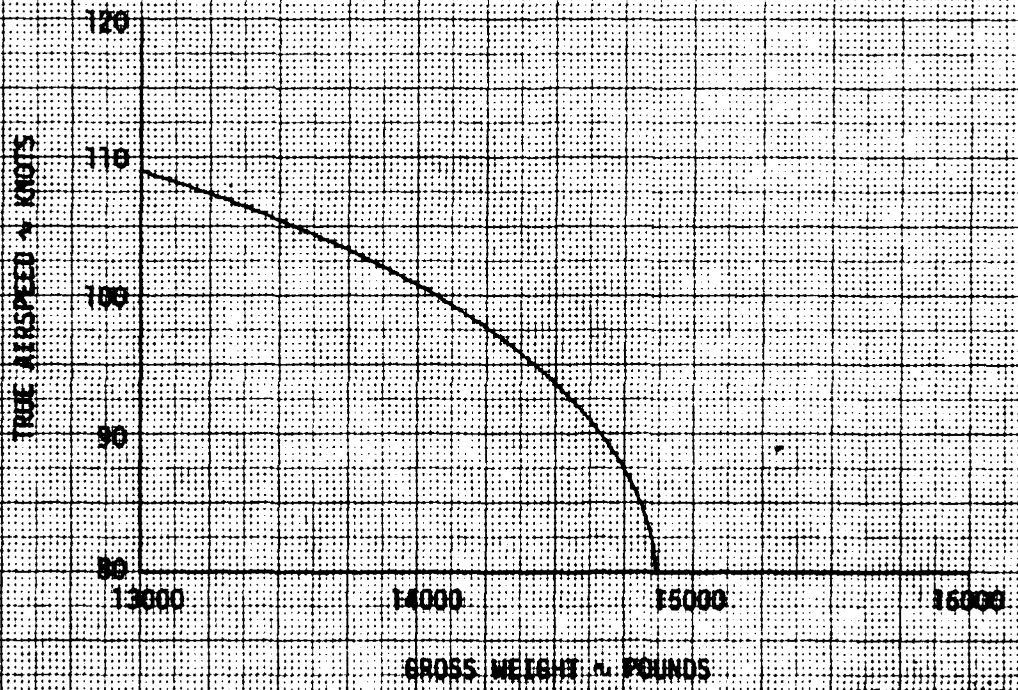


FIGURE 35
LONG-RANGE SUMMARY
YAH-64 USA S/N 74-22249

ROTOR SPEED = 289 RPM
FWD CENTER OF GRAVITY
8-TOW CONFIGURATION

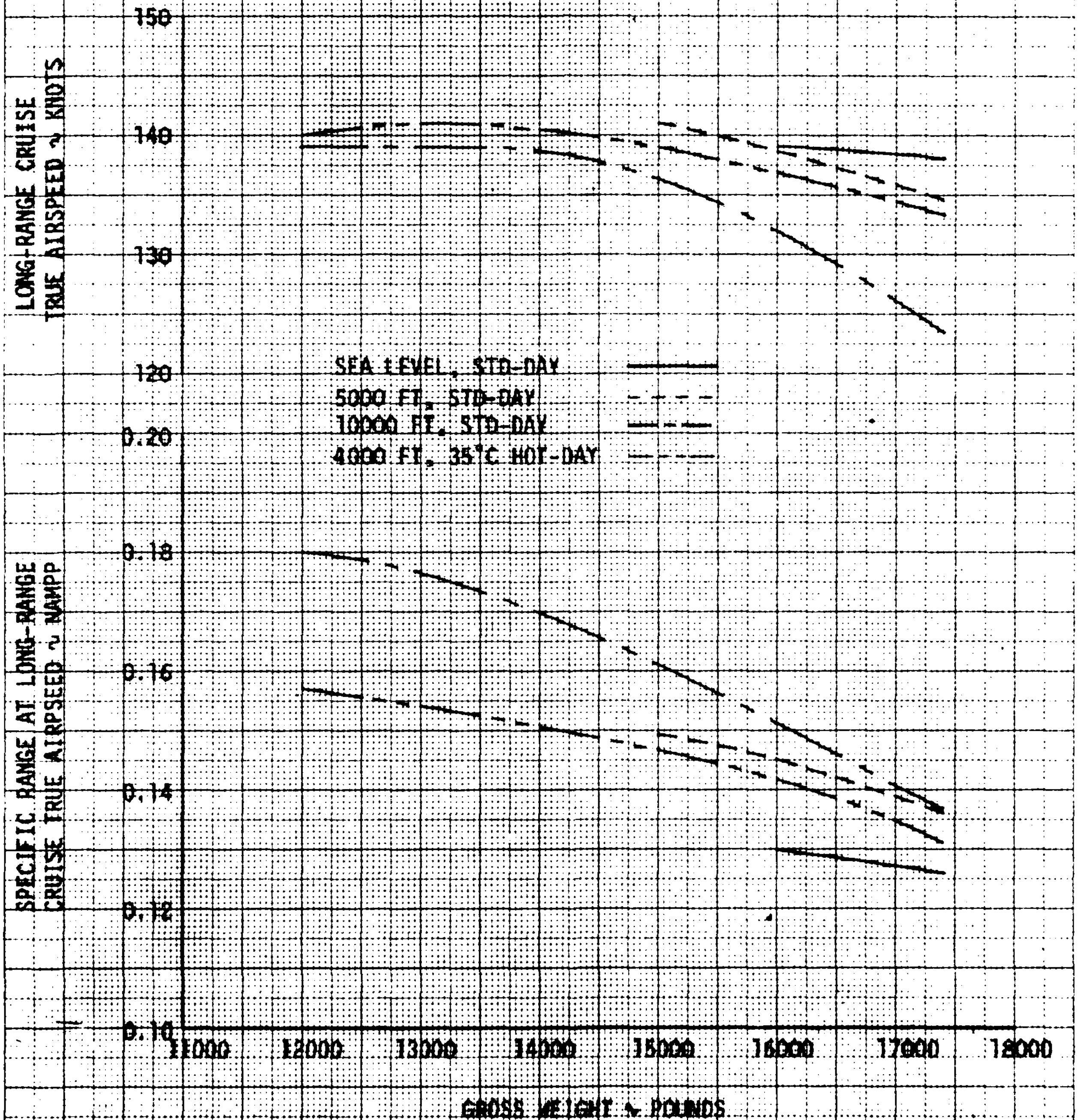


FIGURE 17

MAXIMUM ENDURANCE

YAH-64 USA S/N 74-22249

ROTOR SPEED = 289 RPM

FWD CENTER OF GRAVITY

8-TOW CONFIGURATION

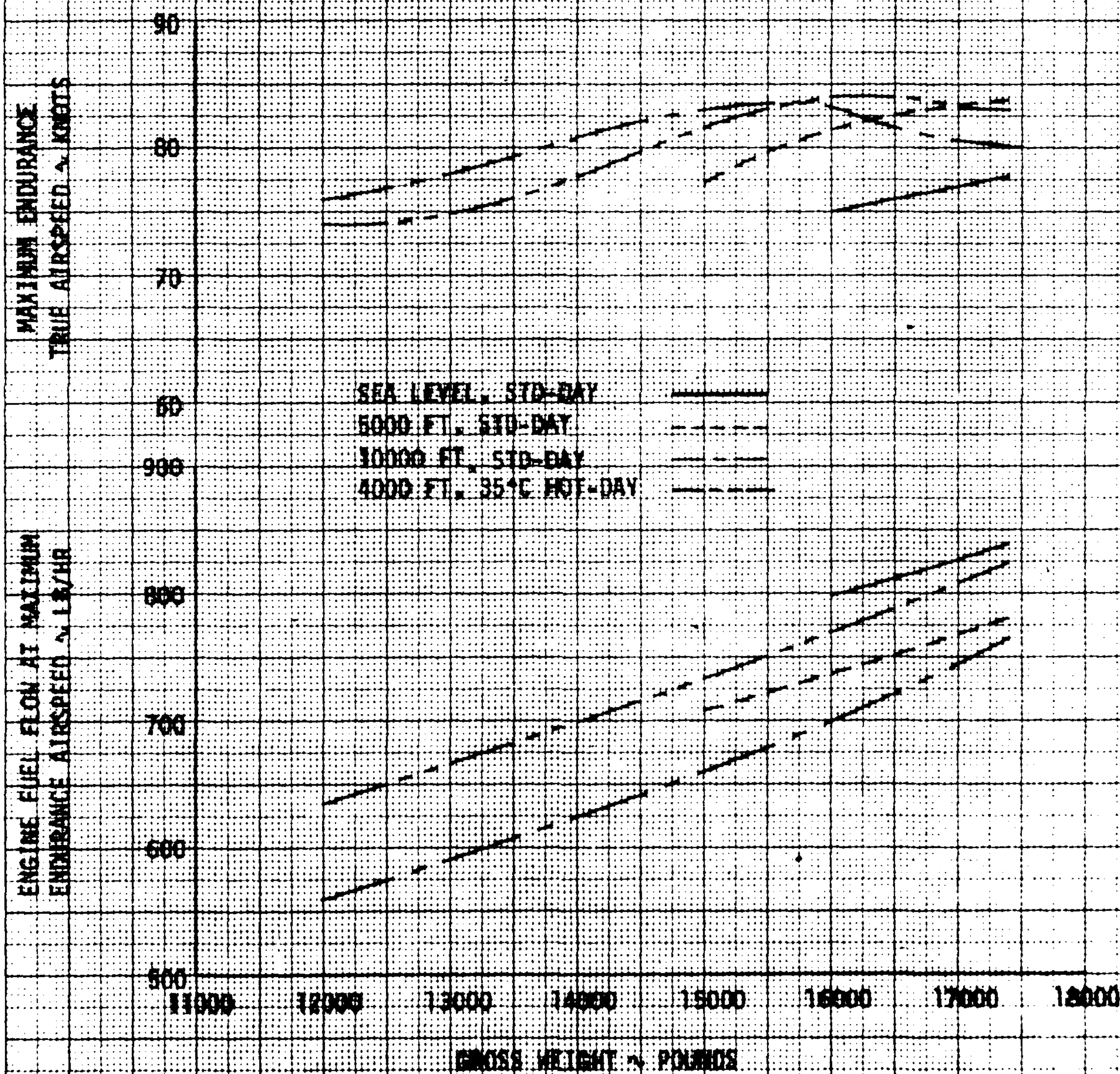


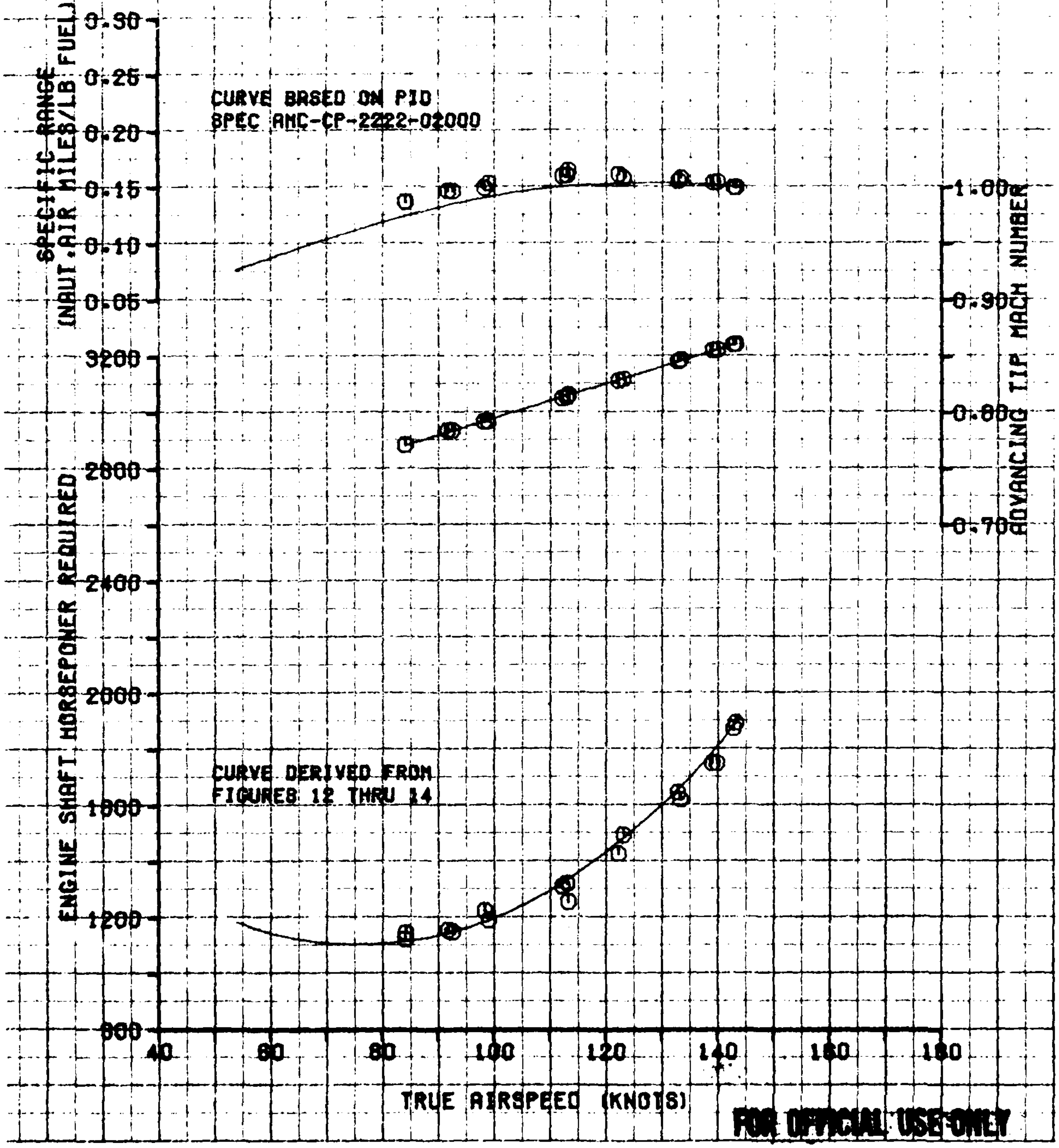
FIGURE 18

LEVEL FLIGHT PERFORMANCE

YAH-64 USA S/N 22249

ENGINE MODEL YT700-GE-700 S/NS 207261.207258

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
14320	199.9(FWD)	.0 MID	4720	20.0	290.0	.007212	8-TOW



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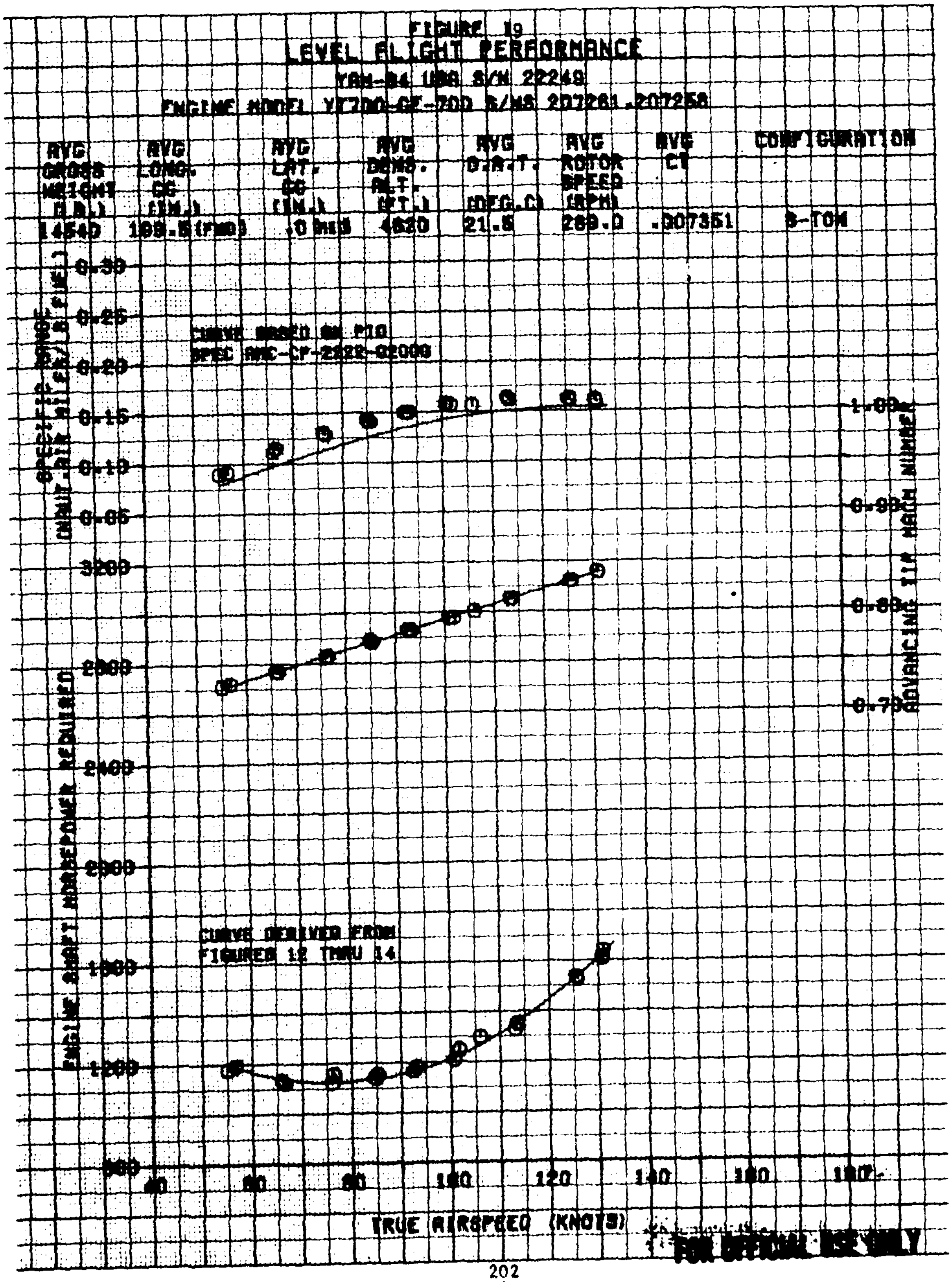
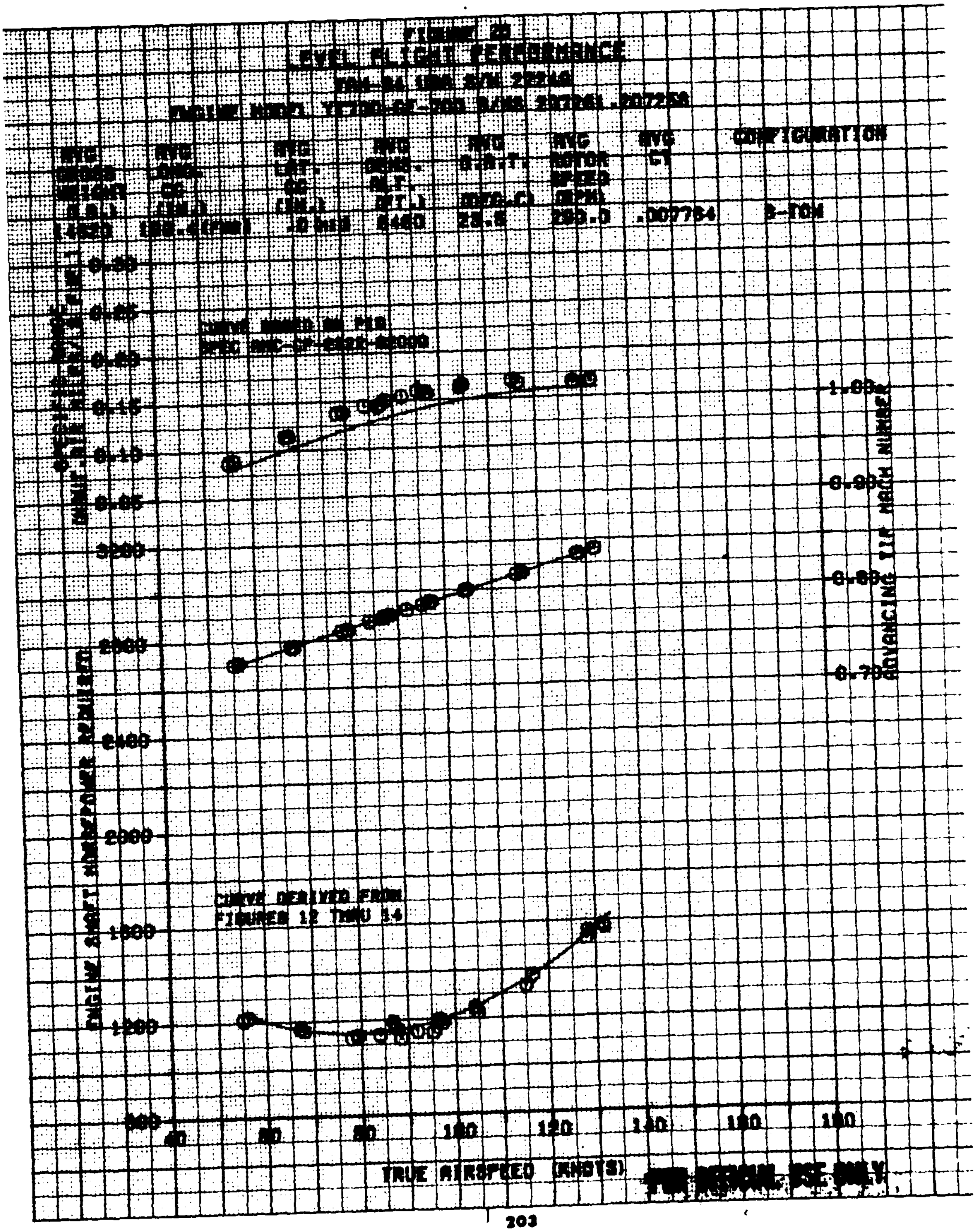
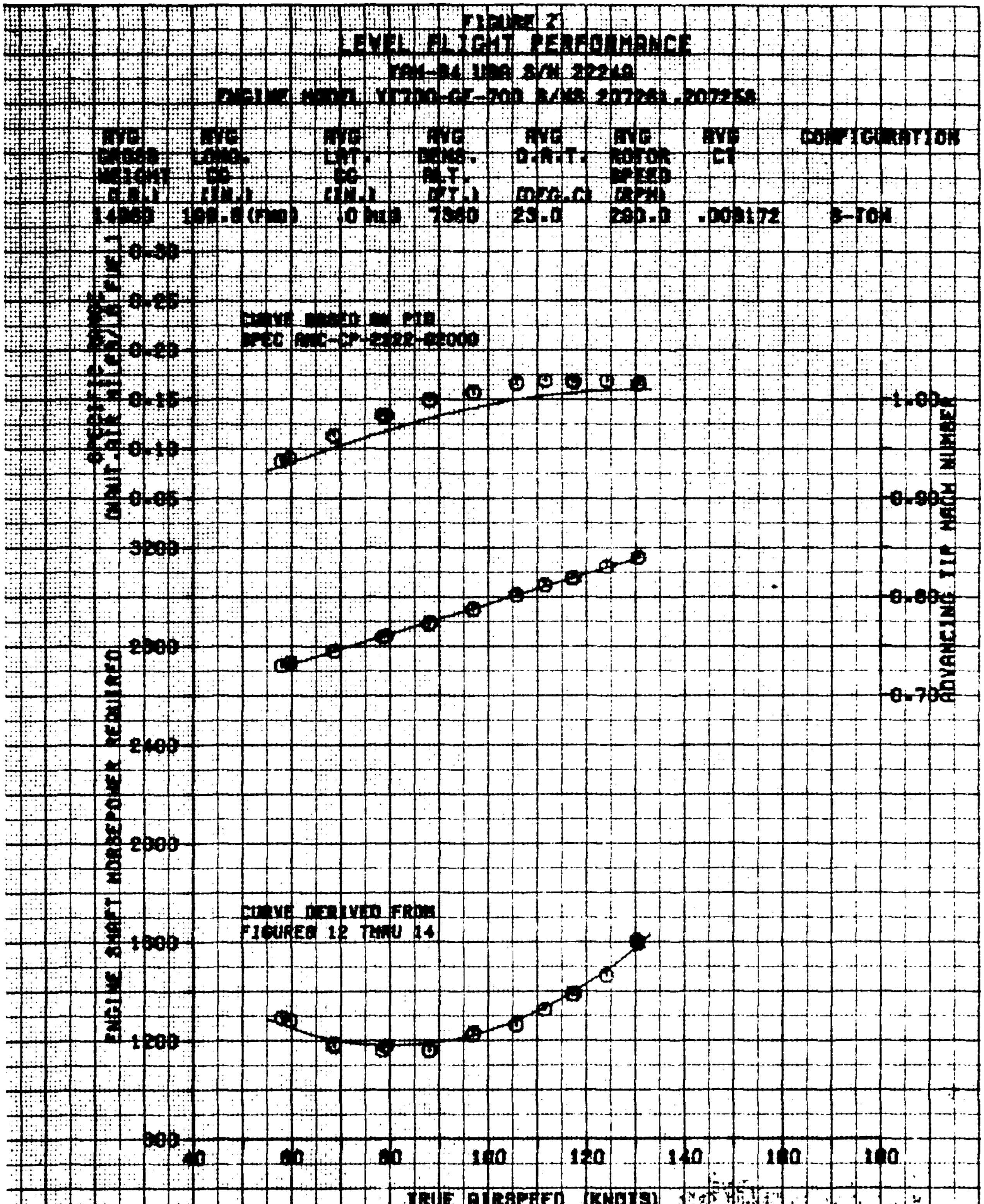


FIGURE 19
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22249
ENGINE MODEL YT700-GE-700 S/NS 207261.207258
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
14540
188.5(FWD)
.0
4620
21.5
289.0
.007351
S-TOW
SPECIFIC RANGE (NAUT.AIR MILES/LB FUEL)
0.30
0.25
0.20
0.15
0.10
0.05
CURVE BASED ON PIO SPEC AMC-CP-2222-02000
1.00
0.90
0.80
0.70
ADVANCING TIP MACH NUMBER
3200
2800
2400
2000
1600
1200
800
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE DERIVED FROM FIGURES 12 THRU 14
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY


LEVEL FLIGHT PERFORMANCE
AVG
CONFIGURATION
ROTOR
SPEED
CT
.007784
CURVE DERIVED FROM
FIGURES 12 THRU 14
ADVANCING TIP MACH NUMBER
1.00
0.90
0.80
0.70
TRUE AIRSPEED (KNOTS)
40
60
80
100
120



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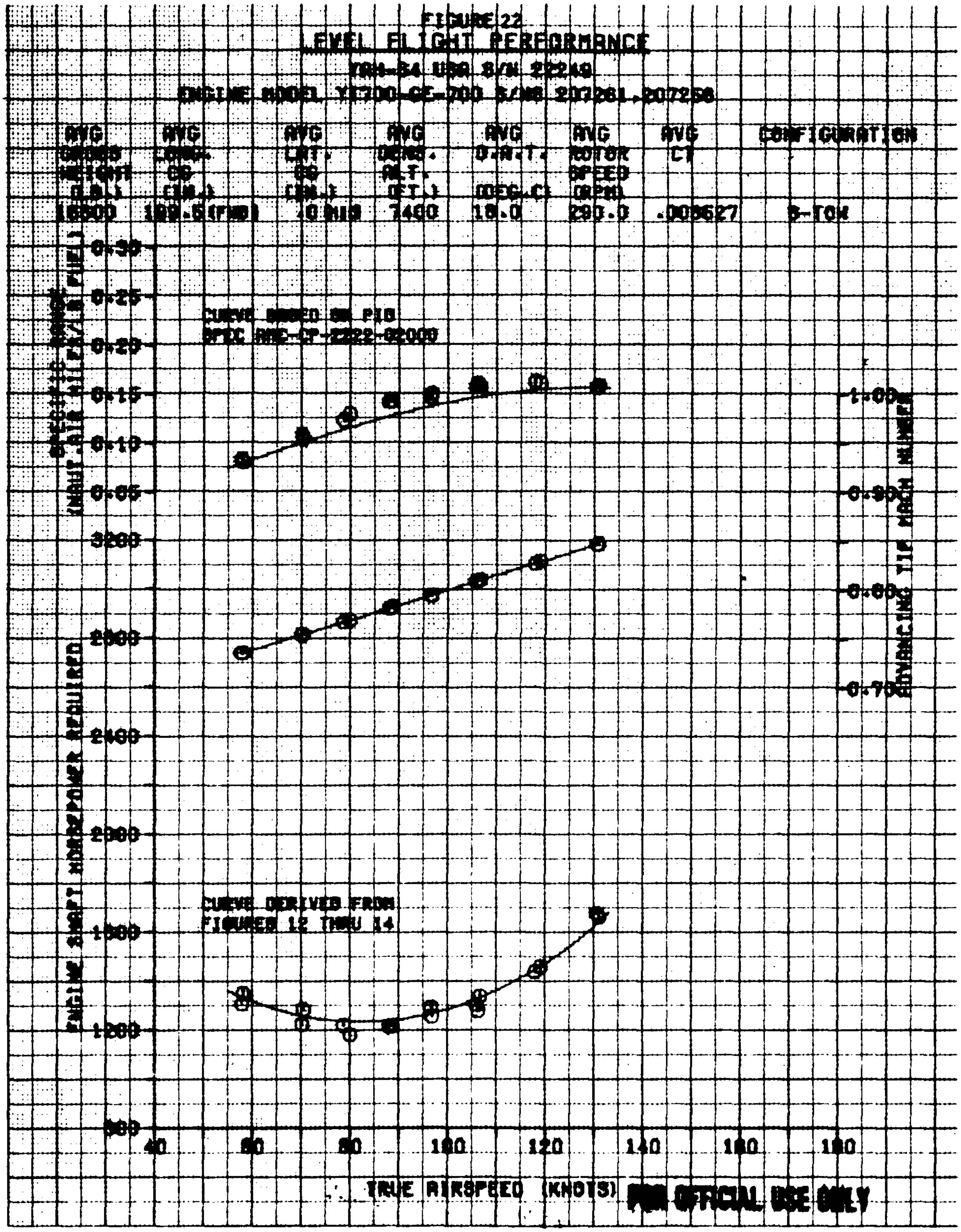


FIGURE 22
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22249
ENGINE MODEL YT700-GE-700 S/NS 207261,207258
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
7400
18.0
290.0
CURVE DERIVED FROM FIGURES 12 THRU 14
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

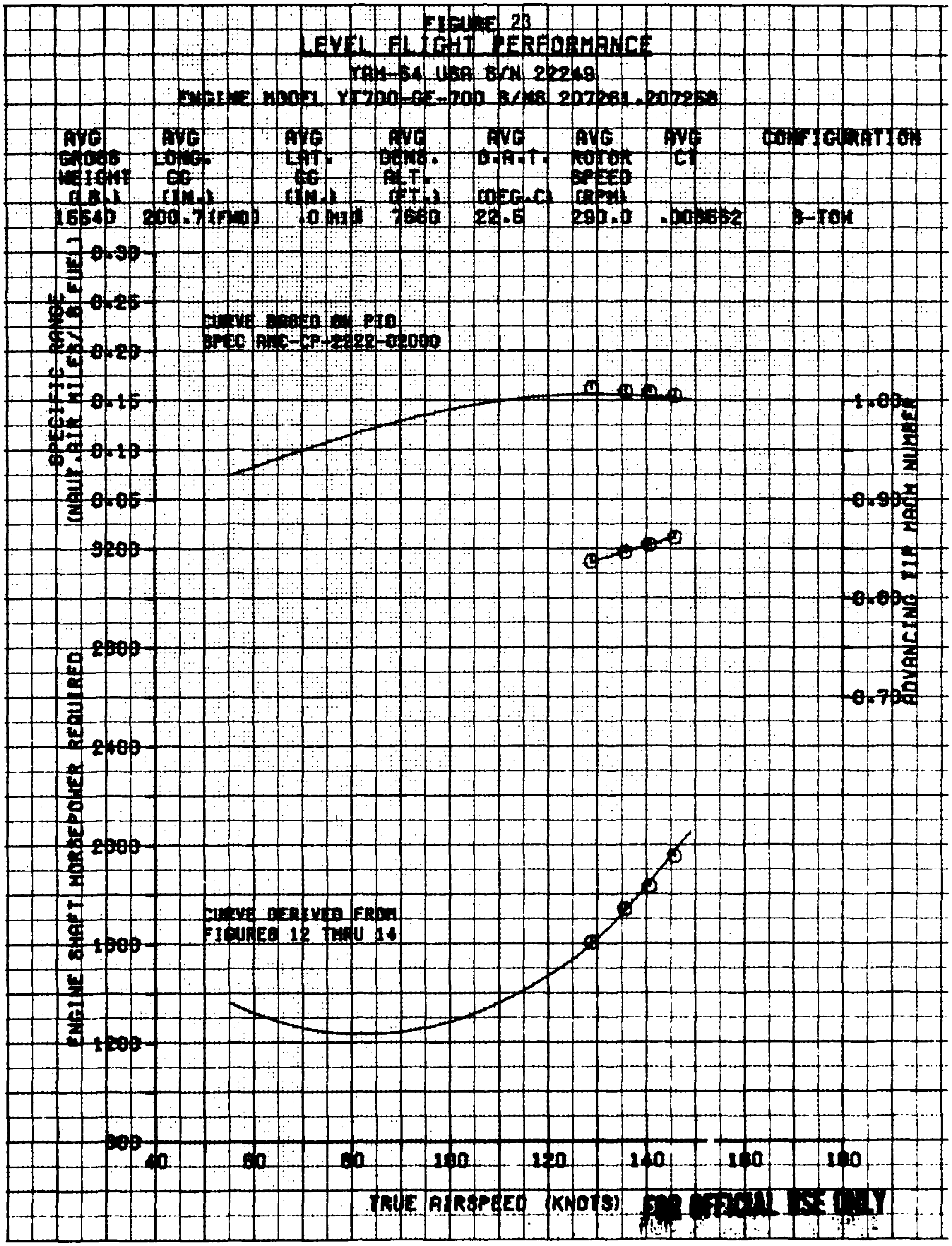
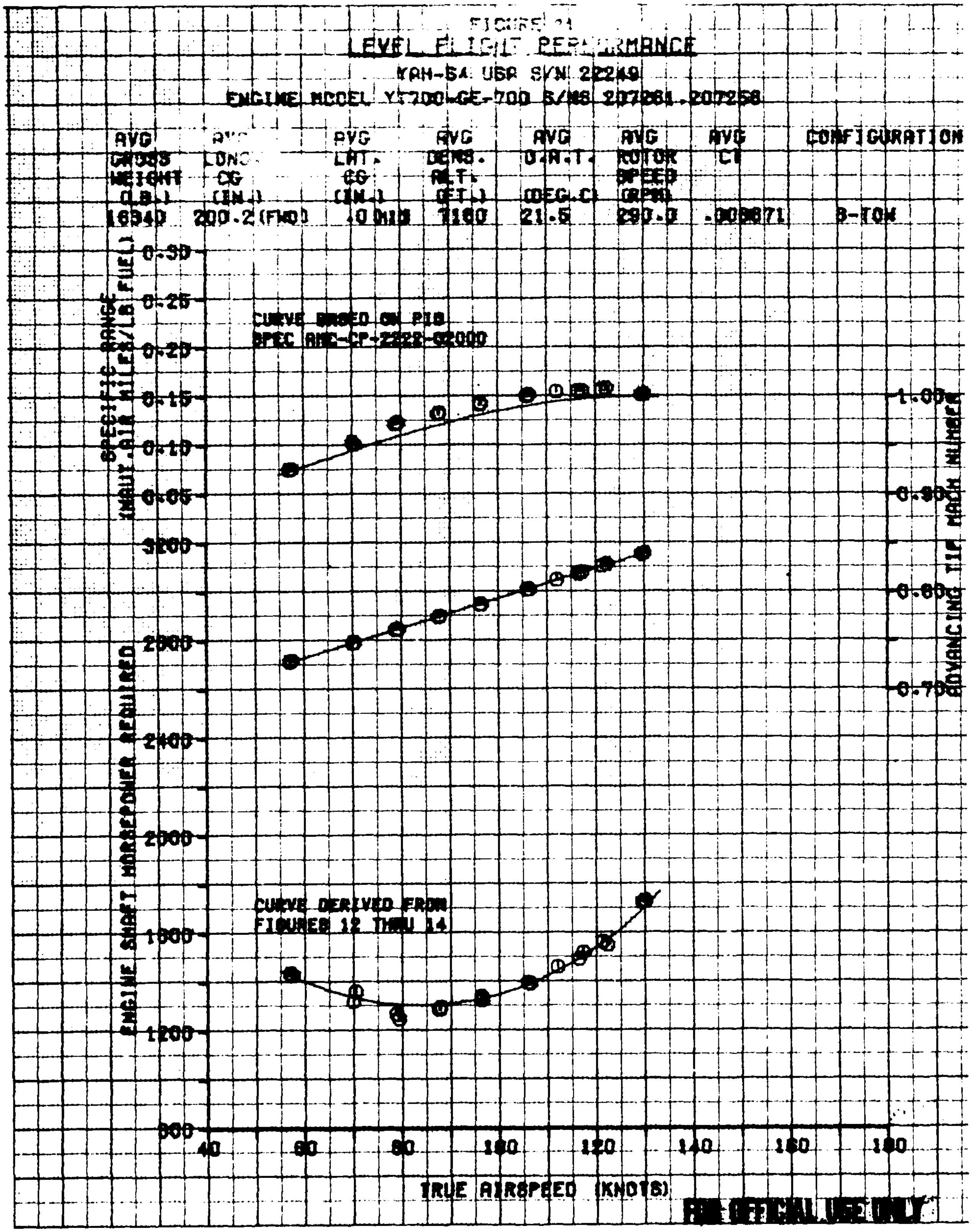
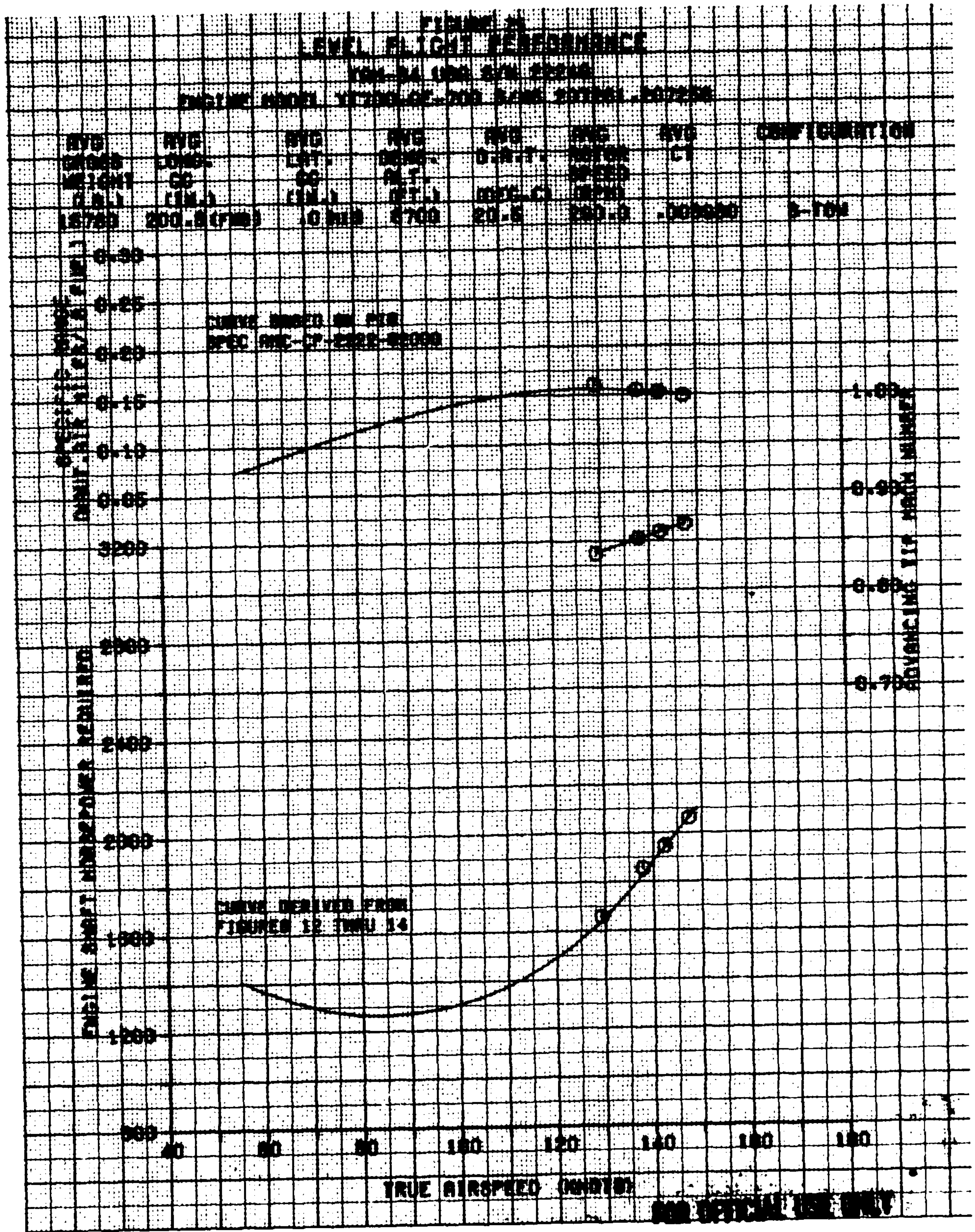
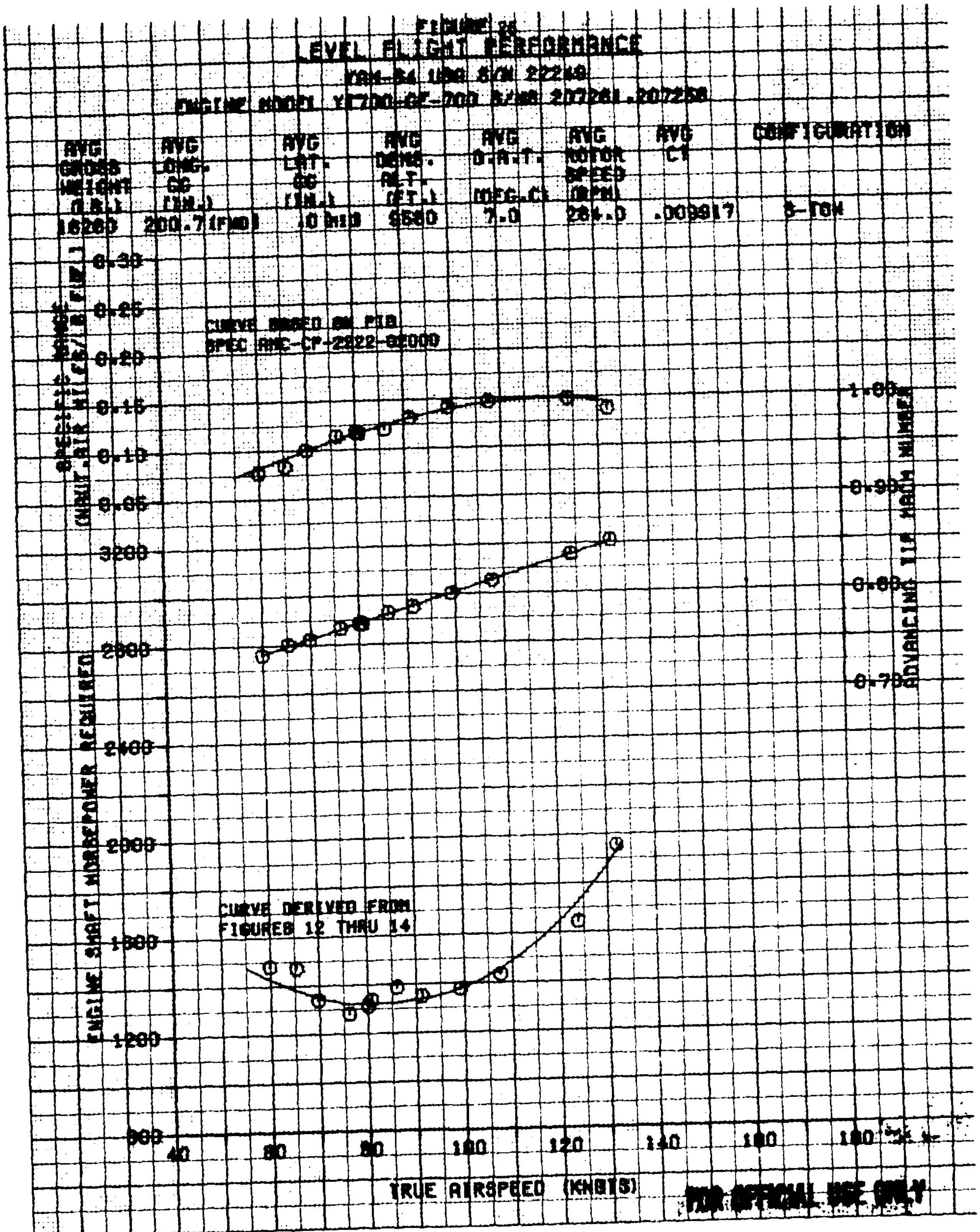


FIGURE 23
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22249
ENGINE MODEL YT700-GE-700 S/NS 207261,207258
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
16540
200.7(FWD)
.0 MID
7660
22.5
290.0
SPECIFIC RANGE (NAUT.AIR MILES/LB FUEL)
CURVE BASED ON PID
SPEC AMC-CP-2222-02000
ADVANCING TIP MACH NUMBER
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE DERIVED FROM
FIGURES 12 THRU 14
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY


LEVEL FLIGHT PERFORMANCE
CONFIGURATION
CURVE DERIVED FROM
FIGURES 12 THRU 14
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)


LEVEL FLIGHT PERFORMANCE
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
200.7 (FWD)
7.0
CURVE BASED ON PIB
CURVE DERIVED FROM
FIGURES 12 THRU 14
ENGINE SHAFT HORSEPOWER REQUIRED
ADVANCING TIP MACH NUMBER
TRUE AIRSPEED (KNOTS)
3200
2800
2400
2000
1600
1200
40
60
80
100
120
140
160
1.00
0.90
0.80
0.70

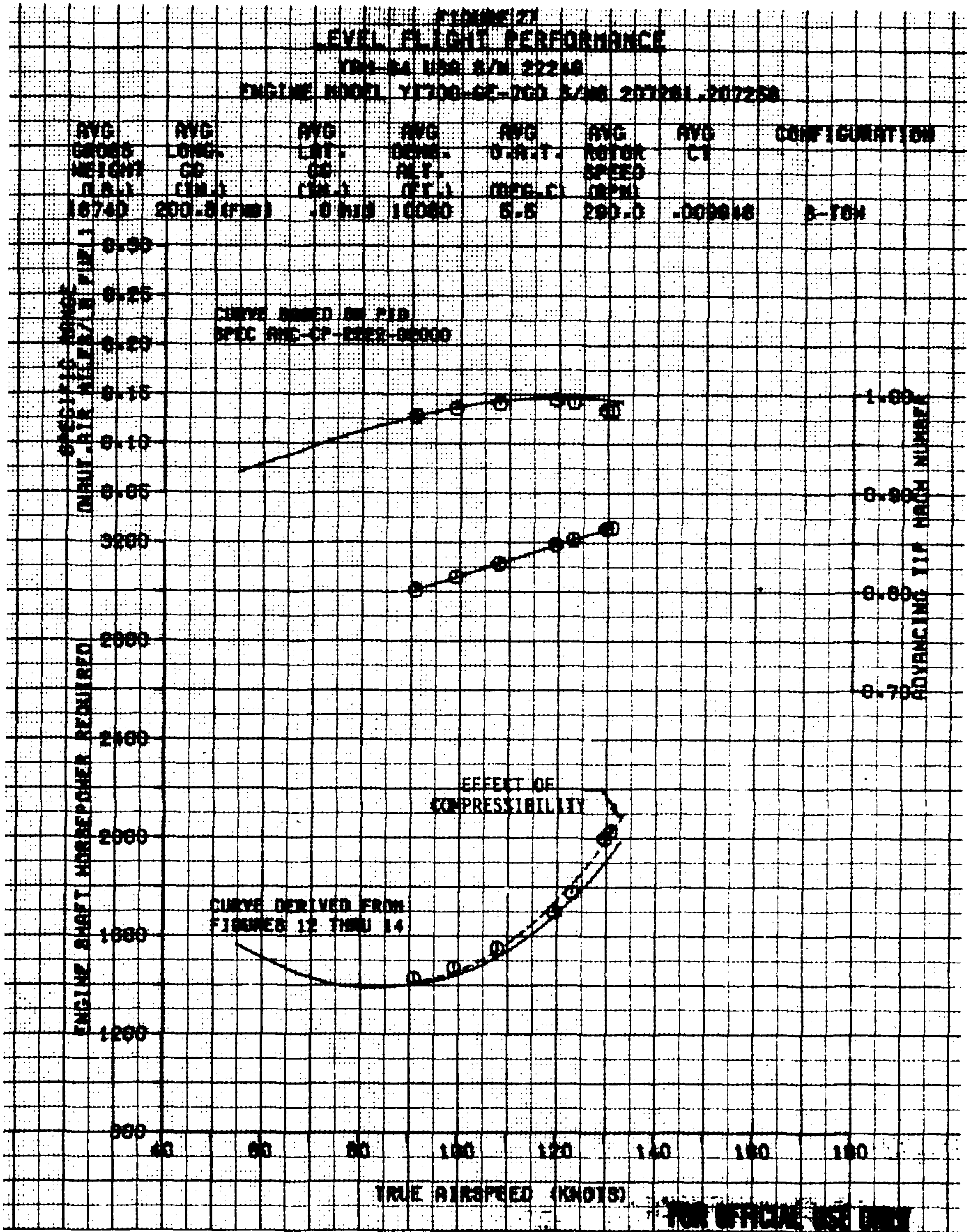
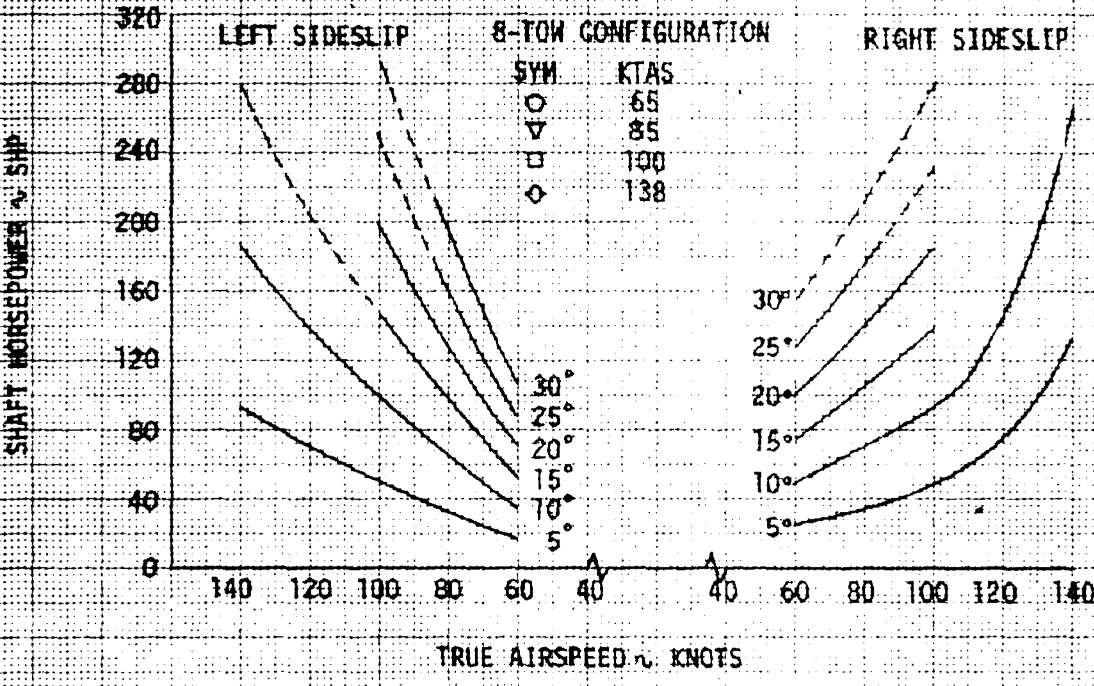
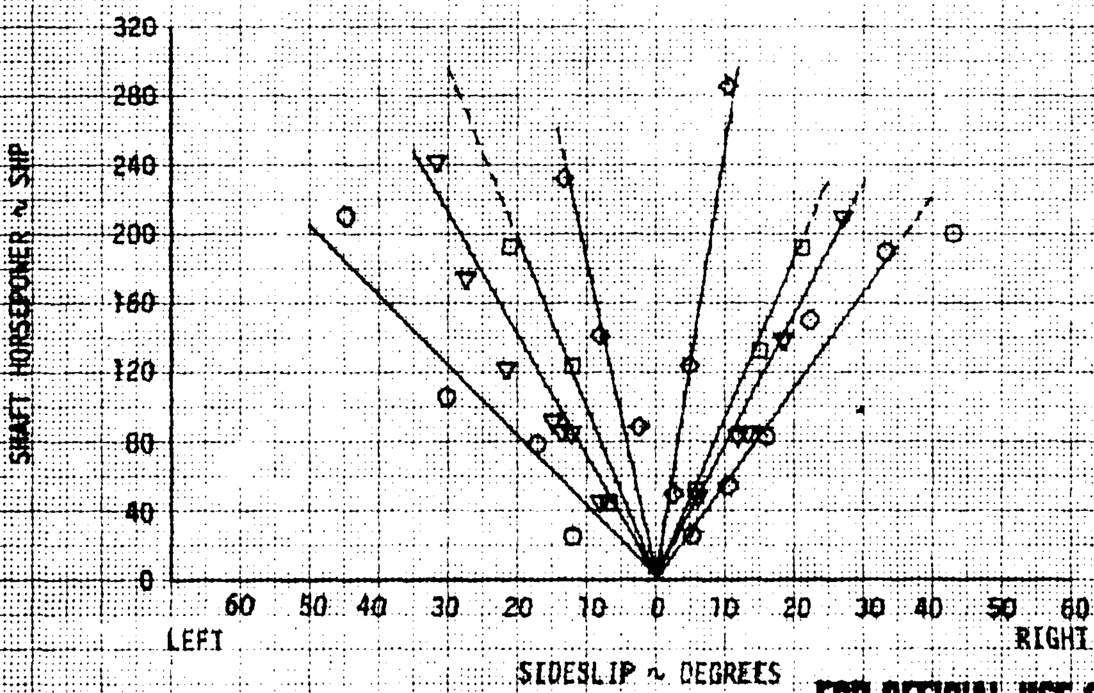


FIGURE 27
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22248
ENGINE MODEL YT700-GE-700 S/NS 207281, 207258
AVG GROSS WEIGHT (LB.) 16740
AVG LONG. CG (IN.) 200.8(FWD)
AVG LAT. CG (IN.) .0 RHS
AVG DENS. ALT. (FT.) 10080
AVG O.A.T. (DEG.C) 6.5
AVG ROTOR SPEED (RPM) 290.0
AVG CT .009846
CONFIGURATION
SPECIFIC RANGE (NAUT.AIR MILES/LB FUEL)
0.30
0.25
0.20
0.15
0.10
0.05
CURVE BASED ON PIB SPEC AMC-CP-2222-02000
ADVANCING TIP MACH NUMBER
1.00
0.90
0.80
0.70
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
EFFECT OF COMPRESSIBILITY
CURVE DERIVED FROM FIGURES 12 THRU 14
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

FIGURE 28
SIDESLIP EFFECTS ON LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 74-22249

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG THRUST COEFFICIENT
14940	200.6 (FWD)	5160	17	289	0.00766





FOR OFFICIAL USE ONLY

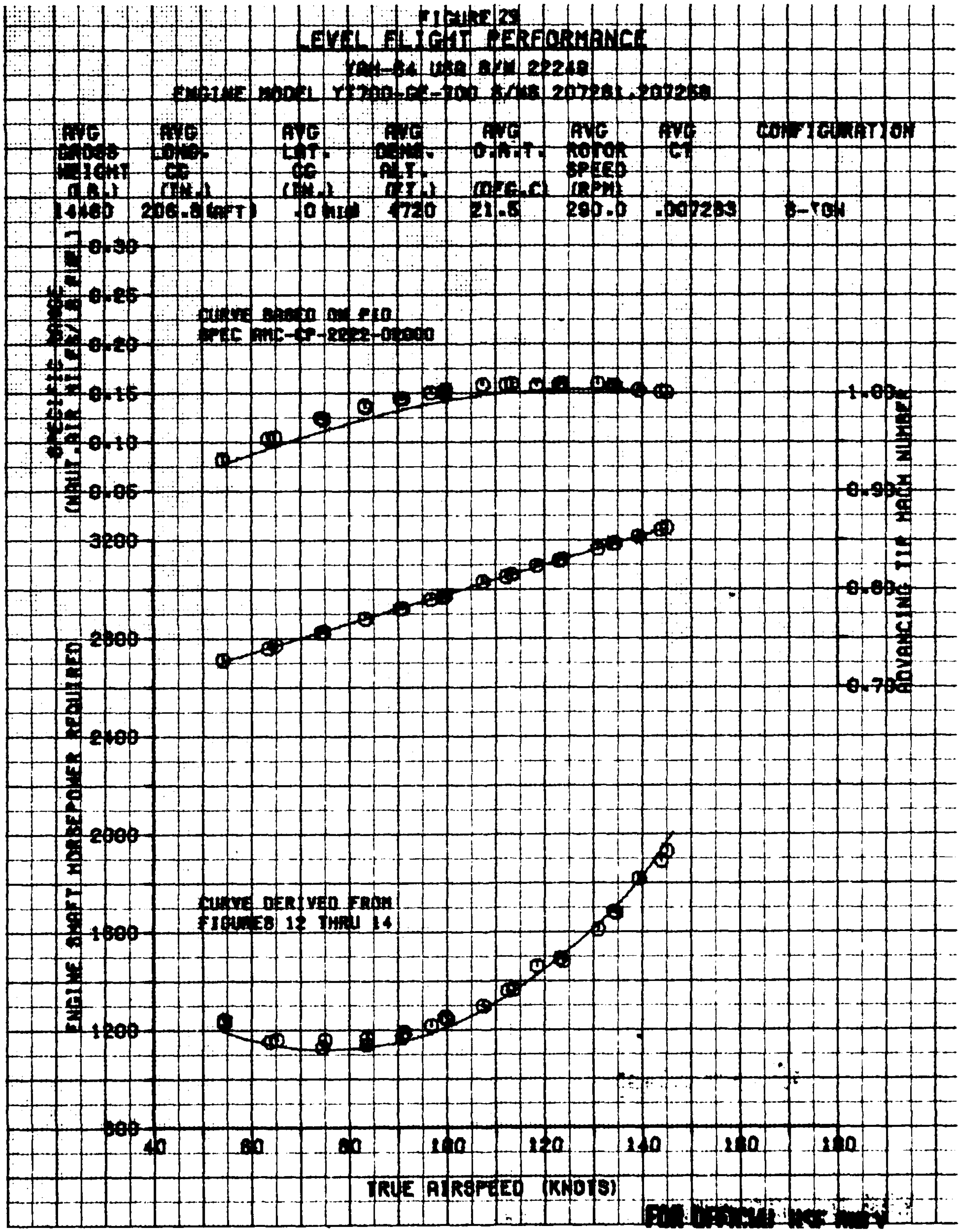


FIGURE 23
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22248
ENGINE MODEL YT700-GE-700 S/NS 207281,207268
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
14480
206.8(AFT)
.0 MID
4720
21.5
290.0
.007283
S-TOW
SPECIFIC RANGE (NAUT.AIR MILES/LB FUEL)
0.30
0.25
0.20
0.15
0.10
0.05
CURVE BASED ON PID
SPEC AMC-CP-2222-02000
ADVANCING TIP MACH NUMBER
1.00
0.90
0.80
0.70
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
CURVE DERIVED FROM
FIGURES 12 THRU 14
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

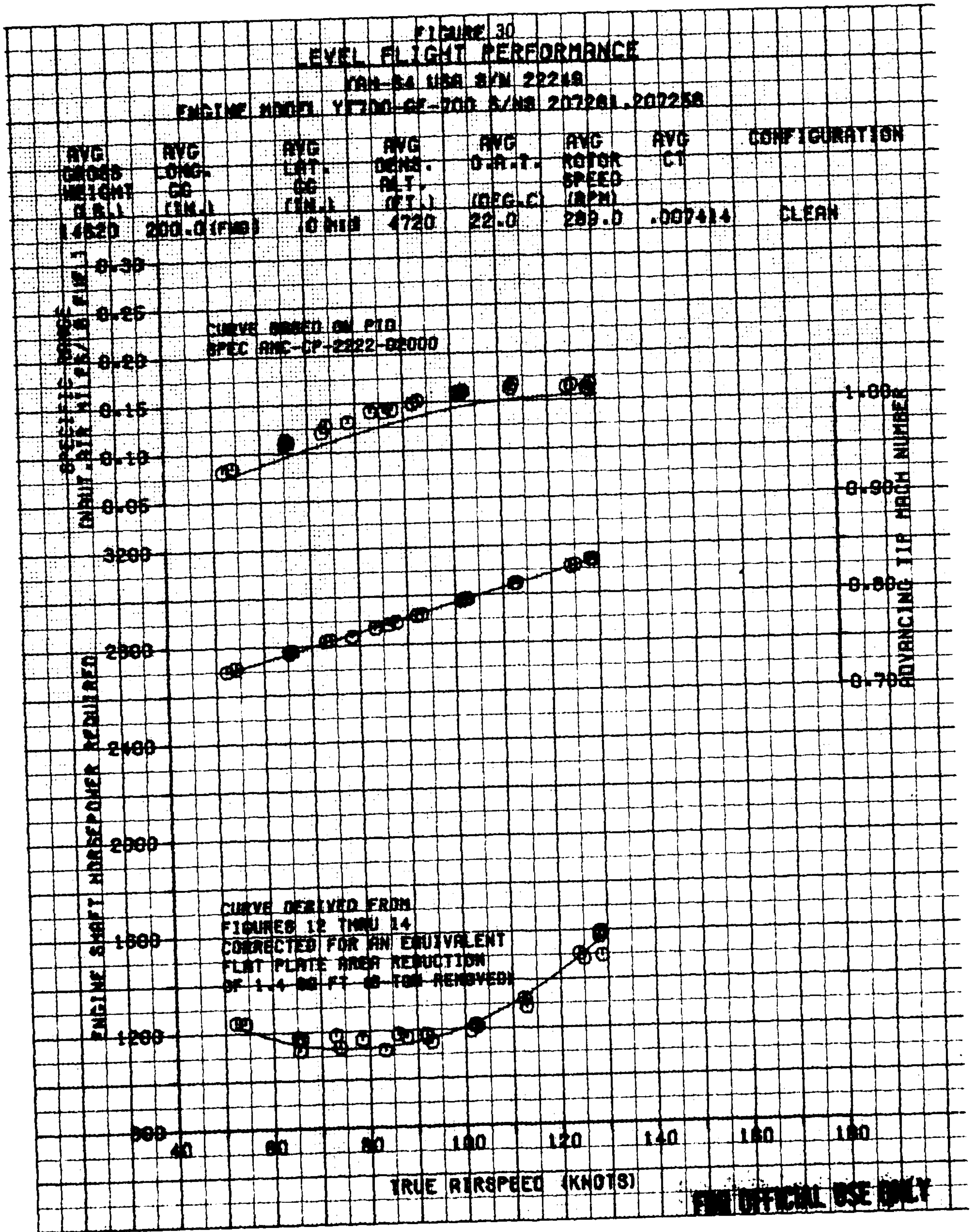


FIGURE 30
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22248
ENGINE MODEL YT700-GE-700 S/NS 207261,207258
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
14620
200.0(FWD)
0.0 MID
4720
22.0
289.0
.007414
CLEAN
SPECIFIC RANGE (NAUT.AIR MI PER/LB FUEL)
0.30
0.25
0.20
0.15
0.10
0.05
CURVE BASED ON PID
SPEC AMC-CP-2222-02000
ADVANCING TIP MACH NUMBER
1.00
0.90
0.80
0.70
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
CURVE DERIVED FROM
FIGURES 12 THRU 14
CORRECTED FOR AN EQUIVALENT
FLAT PLATE AREA REDUCTION
OF 1.4 SQ FT (B-TOW REMOVED)
40
60
80
100
120
140
160
180
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

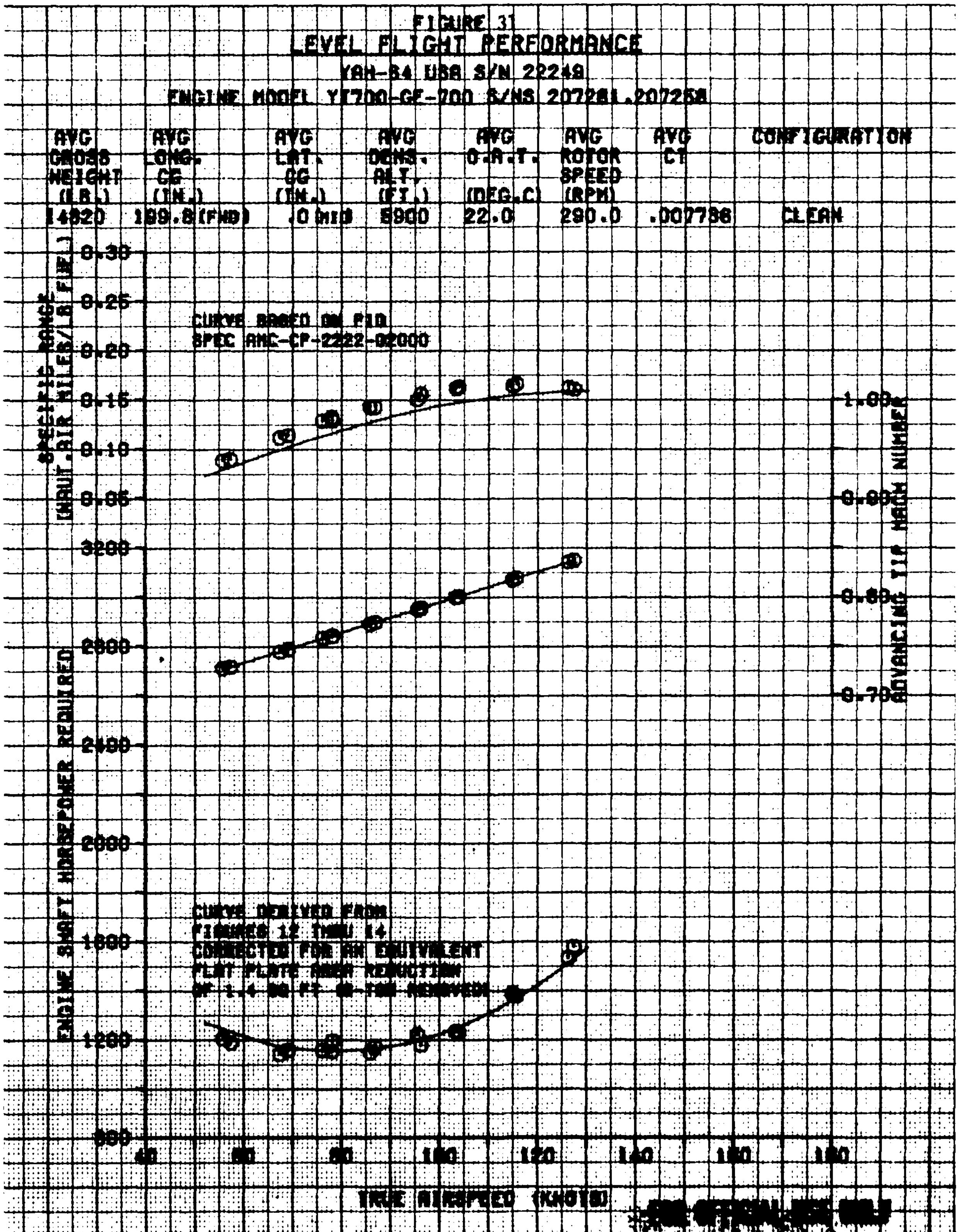


FIGURE 31
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22249
ENGINE MODEL YT700-GE-700 S/NS 207281,207258
AVG GROSS WEIGHT (LB.)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT.)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
14820
199.8(FWD)
.0 MID
5900
22.0
290.0
.007736
CLEAN
SPECIFIC RANGE (NAUT.AIR MILES/LB FUEL)
CURVE BASED ON FID
SPEC AMC-CP-2222-02000
ADVANCING TIP MACH NUMBER
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE DERIVED FROM
FIGURES 12 THRU 14
CORRECTED FOR AN EQUIVALENT
FLAT PLATE AREA REDUCTION
TRUE AIRSPEED (KNOTS)

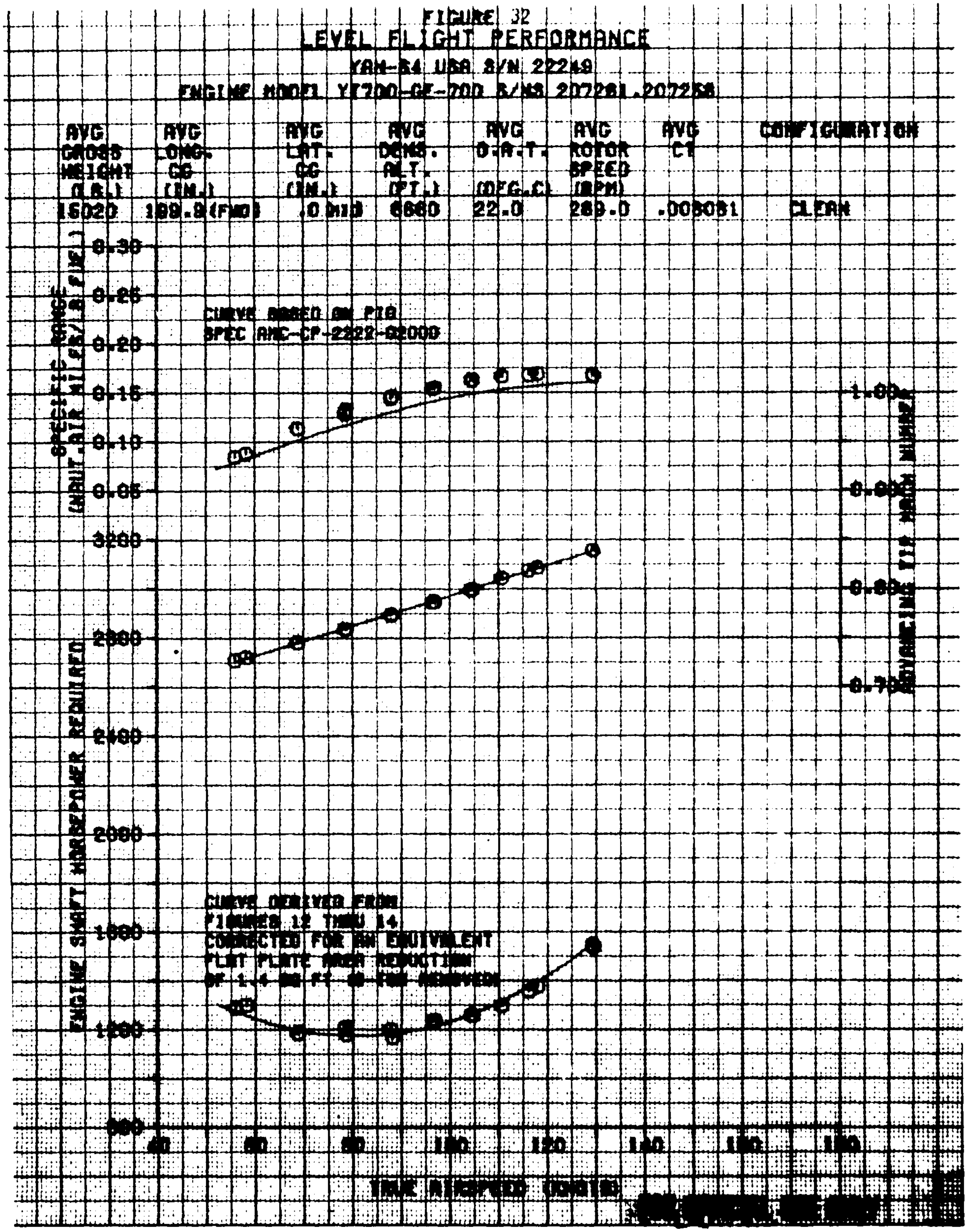


FIGURE 32
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 22249
ENGINE MODEL YT700-GE-700 S/NS 207281,207258
AVG GROSS WEIGHT (LB.) 16020
AVG LONG. CG (IN.) 199.9(FWD)
AVG LAT. CG (IN.) 0.0 MID
AVG DENS. ALT. (FT.) 6860
AVG O.A.T. (DEG.C) 22.0
AVG ROTOR SPEED (RPM) 289.0
AVG CT .008081
CONFIGURATION CLEAN
SPECIFIC RANGE (NAUT.AIR MI/LB FUEL)
CURVE BASED ON PIB SPEC AMC-CP-2222-02000
ENGINE SHAFT HORSEPOWER REQUIRED
CURVE DERIVED FROM FIGURES 12 THRU 14 CORRECTED FOR AN EQUIVALENT FLAT PLATE AREA REDUCTION OF 1.4 SQ FT
ADVANCING TIP MACH NUMBER
TRUE AIRSPEED (KNOTS)

FIGURE 33
VERTICAL DISPLACEMENT
YAH-64 USA S/N 74-22249

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	AVG ROTOR SPEED ~RPM	ENTRY TRUE AIRSPEED ~KT	AVG THRUST COEFFICIENT
14180	203.6(MID)	5500	19.0	289	140	0.00736

NOTES: 1. HORIZONTAL DISTANCE IS DISTANCE REQUIRED TO GAIN 200 FEET IN ALTITUDE FROM ENTRY HEIGHT.
2. AIRSPEED LOSS IS THE CHANGE FROM ENTRY AIRSPEED TO THE AIRSPEED AT THE POINT WHERE THE AIRCRAFT HAS GAINED 200 FEET IN ALTITUDE.

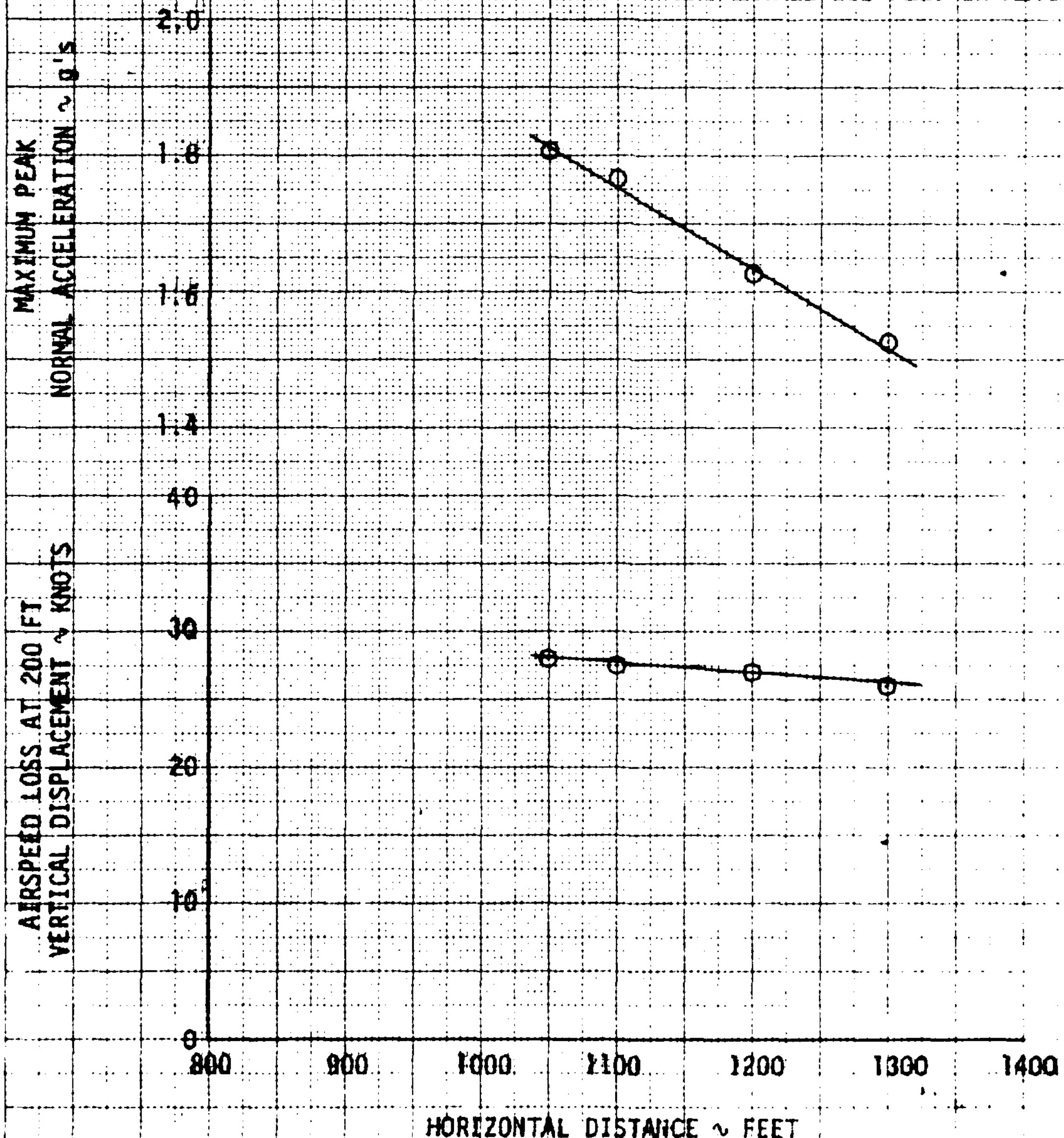


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FIGURE 84
LATERAL ACCELERATION
YAH-64 USA S/N 74-22249

AVG GROSS WEIGHT ~LB	AVG CG LOCATION LONG ~FS	AVG CG LOCATION LAT ~BL	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	AVG ROTOR SPEED ~RPM	CONFIGURATION
14200	203.4(MID)	.0(MID)	5440	19	289	8-TOW

NOTE: RIGHT LATERAL ACCELERATIONS ONLY

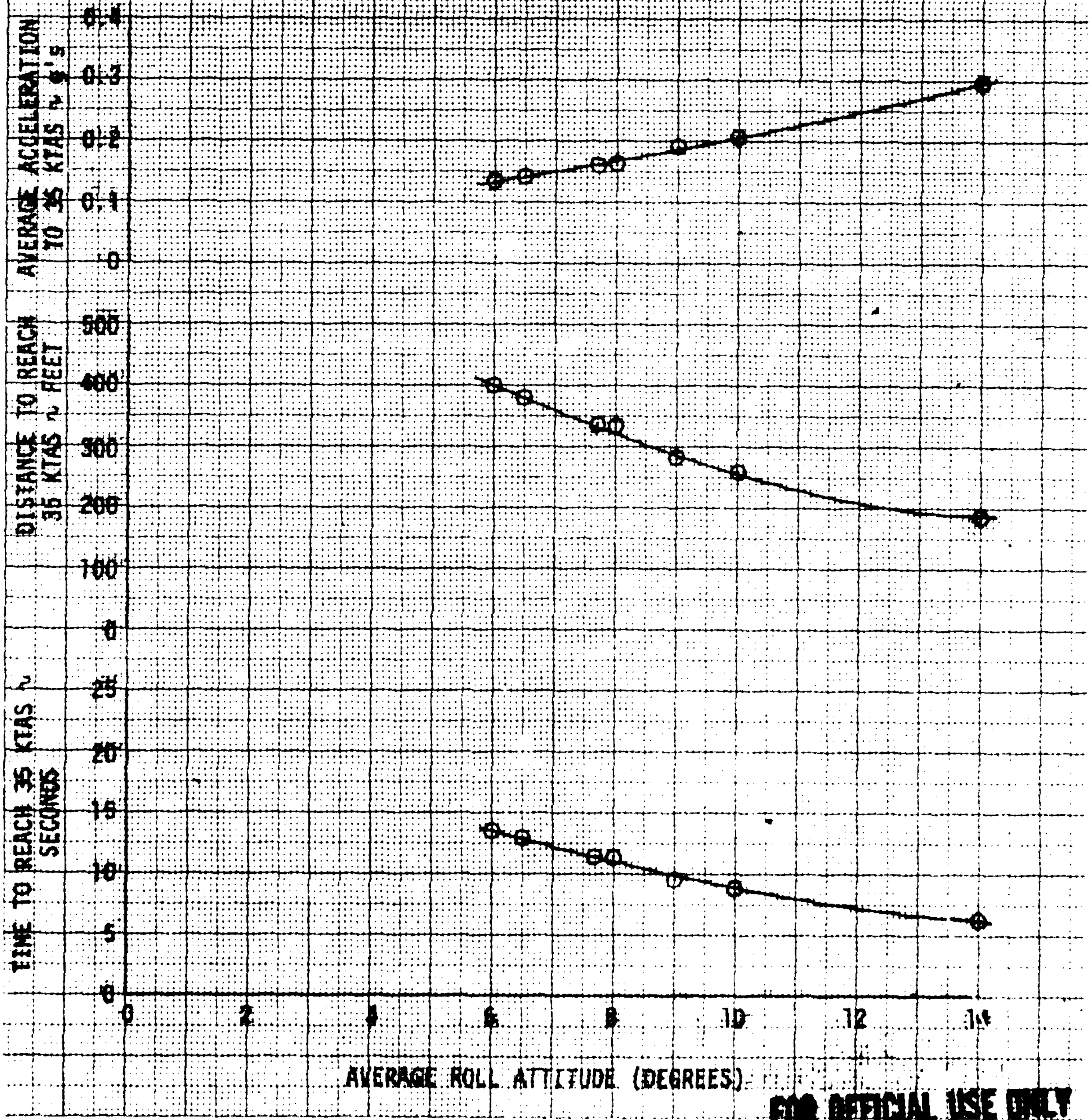

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FIGURE 35

AUTOROTATIONAL DESCENT PERFORMANCE

YAH-64 S/N 74-22249

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	AVG ROTOR SPEED ~RPM	AVG THRUST COEFFICIENT	CONFIGURATION
14200	206.4 (AFT)	5000	18.0	289	0.00726	8-TOW

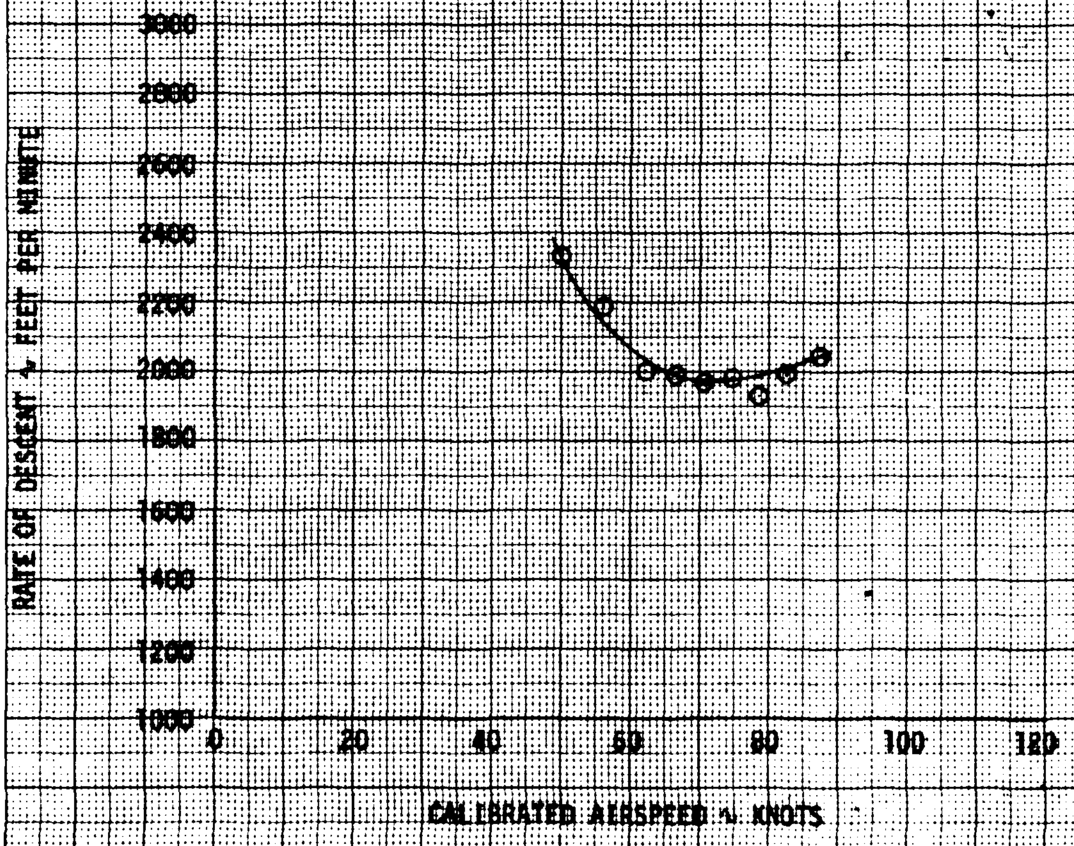

FIGURE 36
AUTOROTATIONAL DESCENT PERFORMANCE
YAH-64 S/N 77-22248

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	AVG CALIBRATED AIRSPEED ~ KT	AVG THRUST COEFFICIENT	CONFIGURATION
13700	206.0 (AFT)	5850	18	75	0.00748	8-TOW

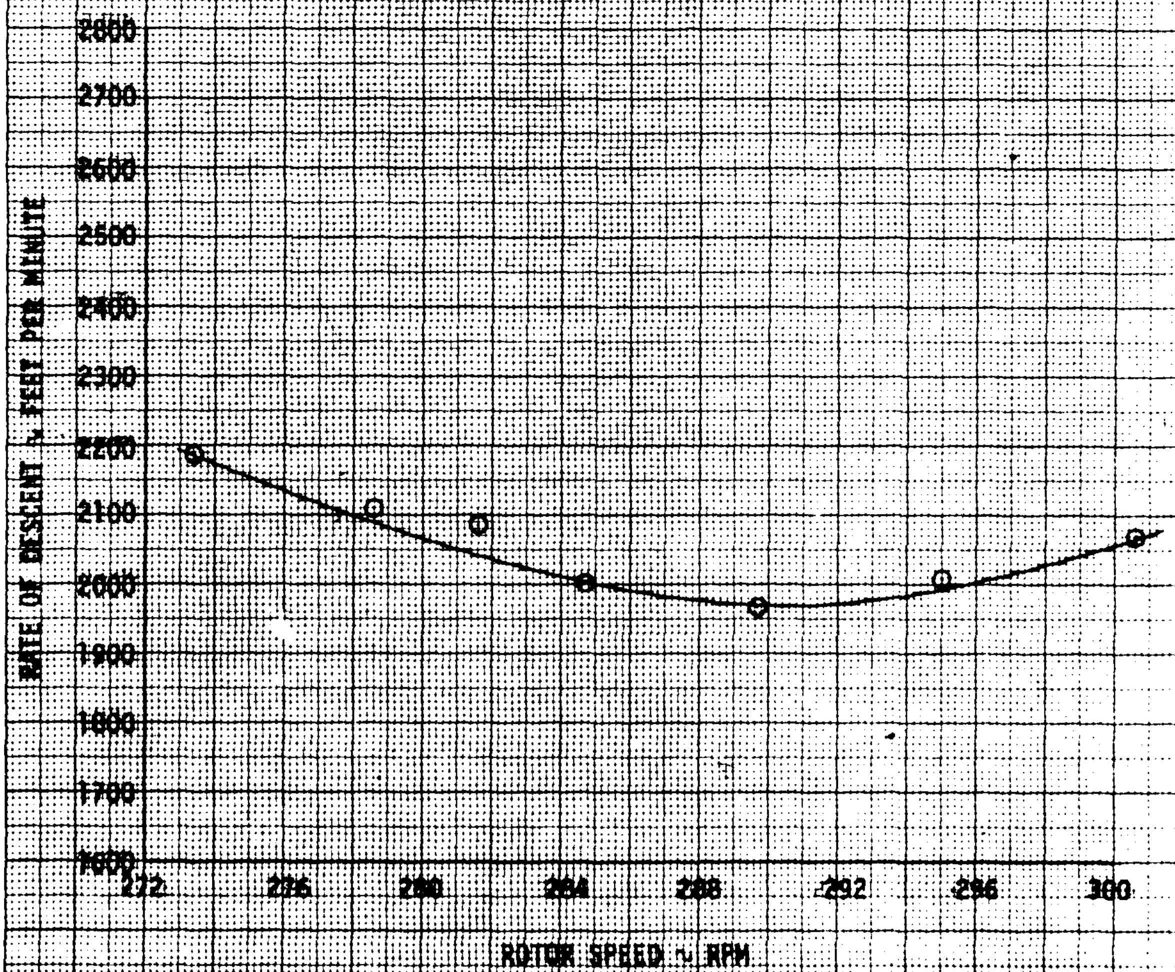

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FIGURE 37

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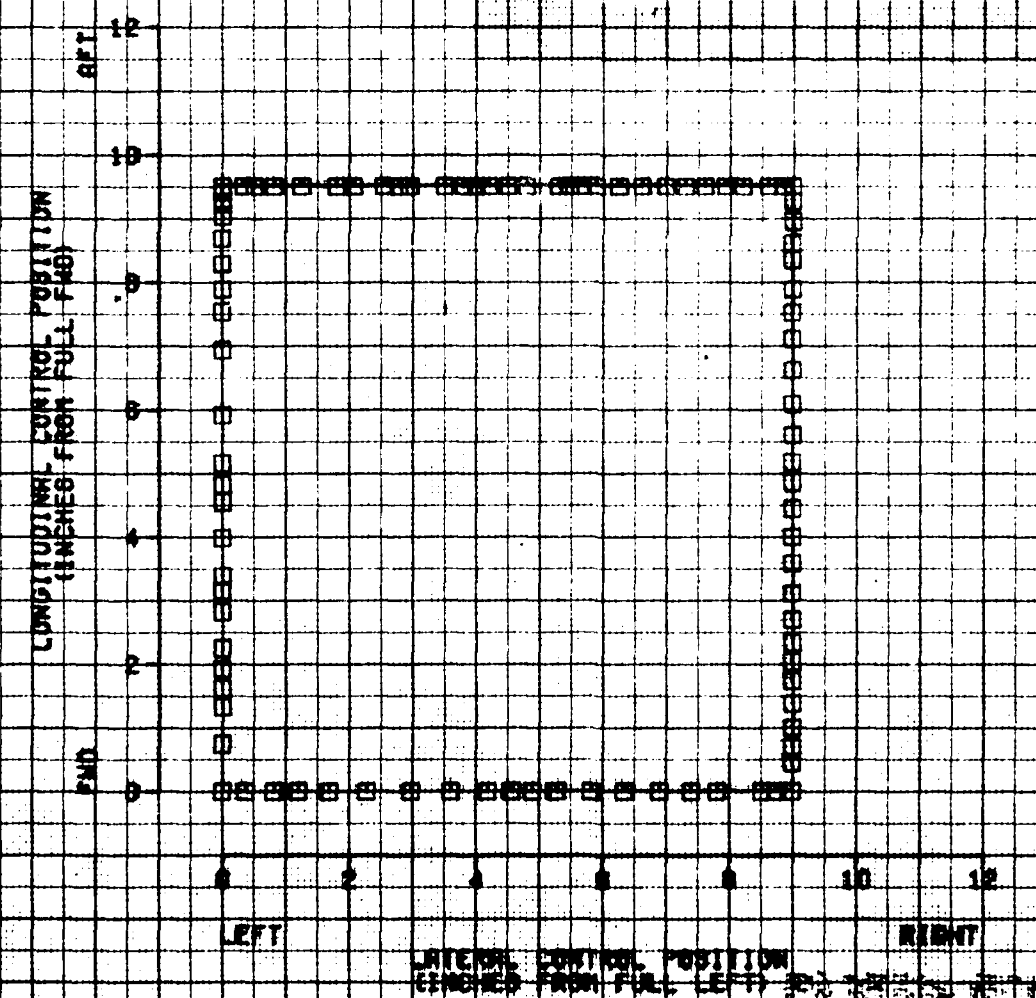
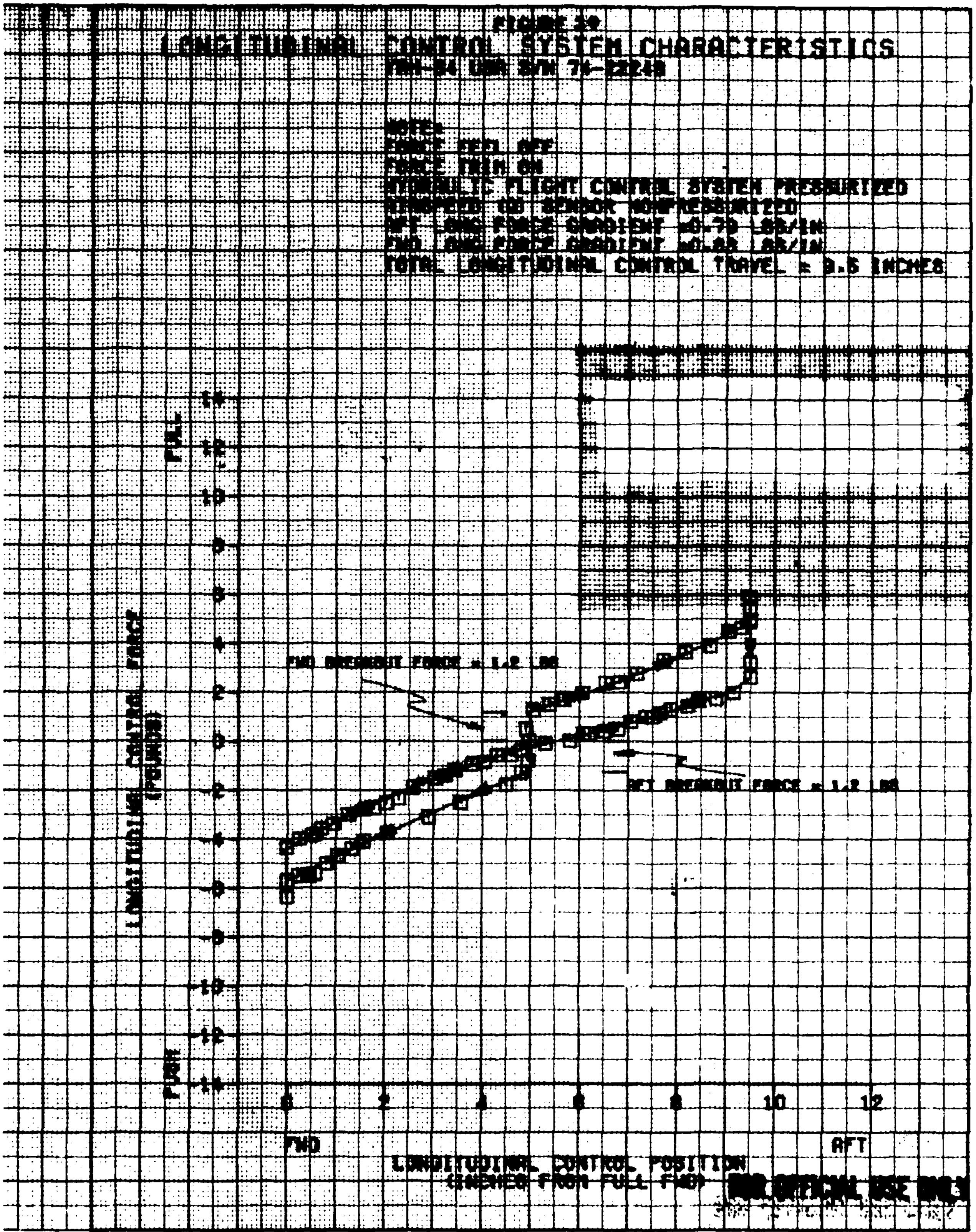
FIGURE 38

LIMITS OF CYCLIC CONTROL TRAVEL

YAH-64 USA S/N 74-22248

NOTE:
HYDRAULIC FLIGHT CONTROL SYSTEM PRESSURIZED
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.5 INCHES
TOTAL LATERAL CONTROL TRAVEL = 9.0 INCHES





LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
HYDRAULIC FLIGHT CONTROL SYSTEM PRESSURIZED
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.5 INCHES
FWD BREAKOUT FORCE = 1.2 LBS
AFT BREAKOUT FORCE = 1.2 LBS
LONGITUDINAL CONTROL FORCE
(POUNDS)
PULL
PUSH
0
2
4
6
8
10
12
FWD
AFT
LONGITUDINAL CONTROL POSITION
(INCHES FROM FULL FWD)
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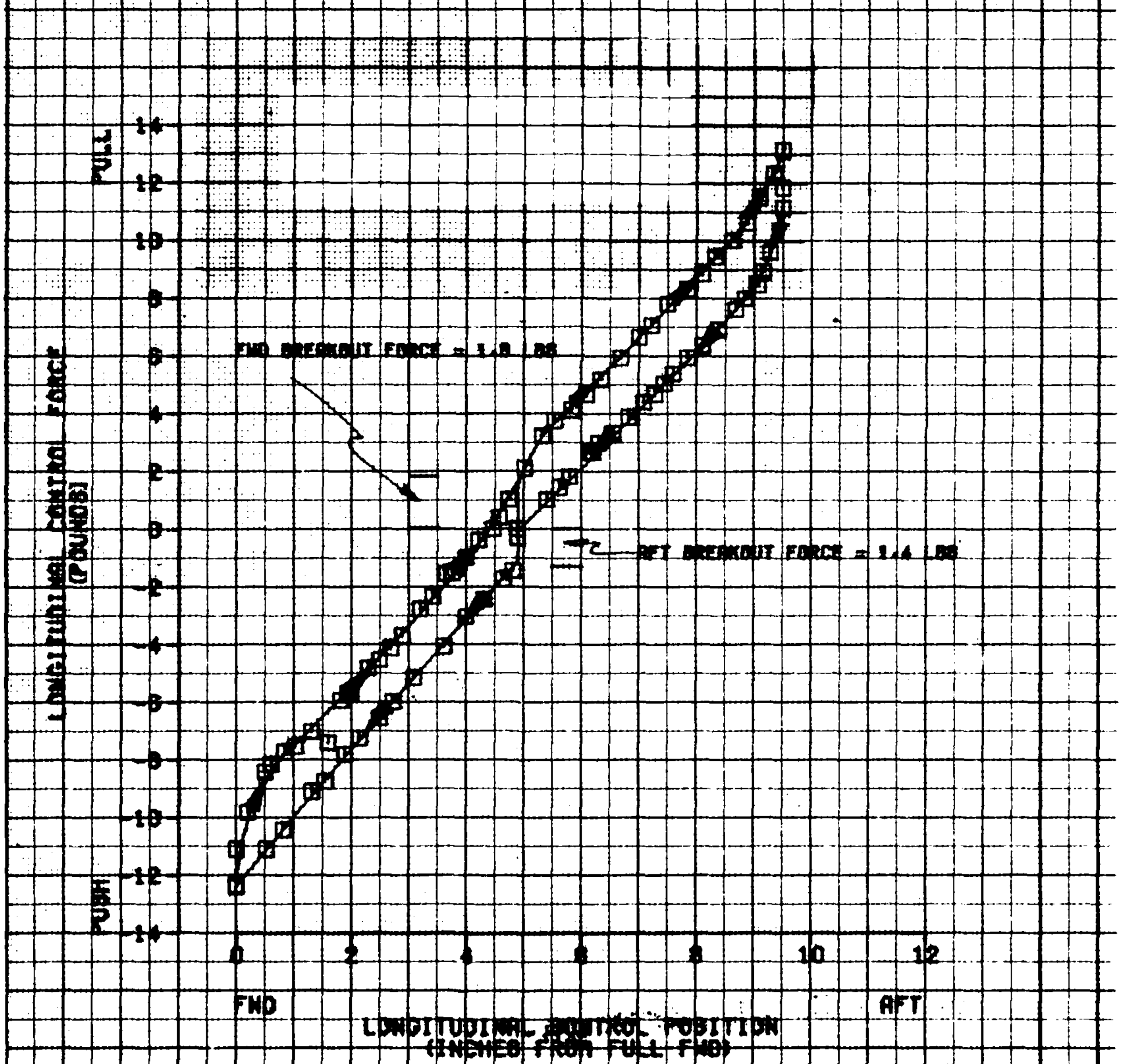
FIGURE 49

LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS

YAH-64 USA S/N 74-22248

NOTE:
FORCE FEEL ON
FORCE TRIM ON
HYDRAULIC FLIGHT CONTROL SYSTEM PRESSURIZED
AIRSPEED (Q) SENSOR NONPRESSURIZED
AFT LONG FORCE GRADIENT =2.1 LBS/IN
FWD LONG FORCE GRADIENT =2.2 LBS/IN
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.5 INCHES



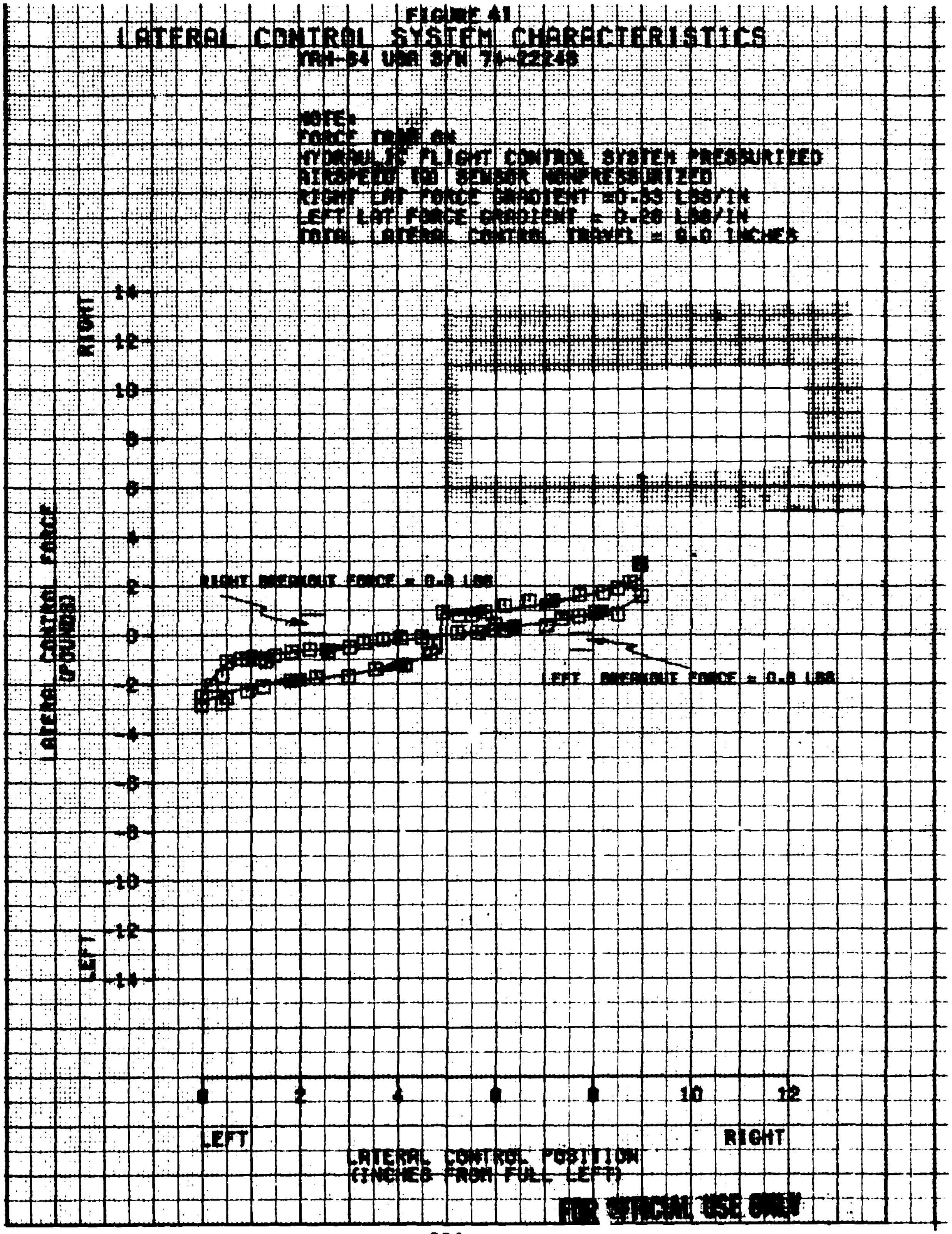


FIGURE 41
LATERAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 74-22248
NOTE:
FORCE TRIM ON
HYDRAULIC FLIGHT CONTROL SYSTEM PRESSURIZED
AIRSPEED (Q) SENSOR NONPRESSURIZED
RIGHT LAT FORCE GRADIENT =0.33 LBS/IN
LEFT LAT FORCE GRADIENT = 0.28 LBS/IN
TOTAL LATERAL CONTROL TRAVEL = 9.0 INCHES
RIGHT BREAKOUT FORCE = 0.8 LBS
LEFT BREAKOUT FORCE = 0.8 LBS
RIGHT
LEFT
LATERAL CONTROL FORCE (POUNDS)
14
12
10
8
6
4
2
0
-2
-4
-6
-8
-10
-12
-14
0
2
4
6
8
10
12
LEFT
RIGHT
LATERAL CONTROL POSITION (INCHES FROM FULL LEFT)

FIGURE 42
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 74-22248

NOTE:
FORCE TRIM ON
HYDRAULIC FLIGHT CONTROL SYSTEM PRESSURIZED
AIRSPEED (Q) SENSOR NONPRESSURIZED
DIRECTIONAL FORCE GRADIENT = 2.9 LBS/IN. RT
DIRECTIONAL FORCE GRADIENT = 2.6 LBS/IN. LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.9 INCHES

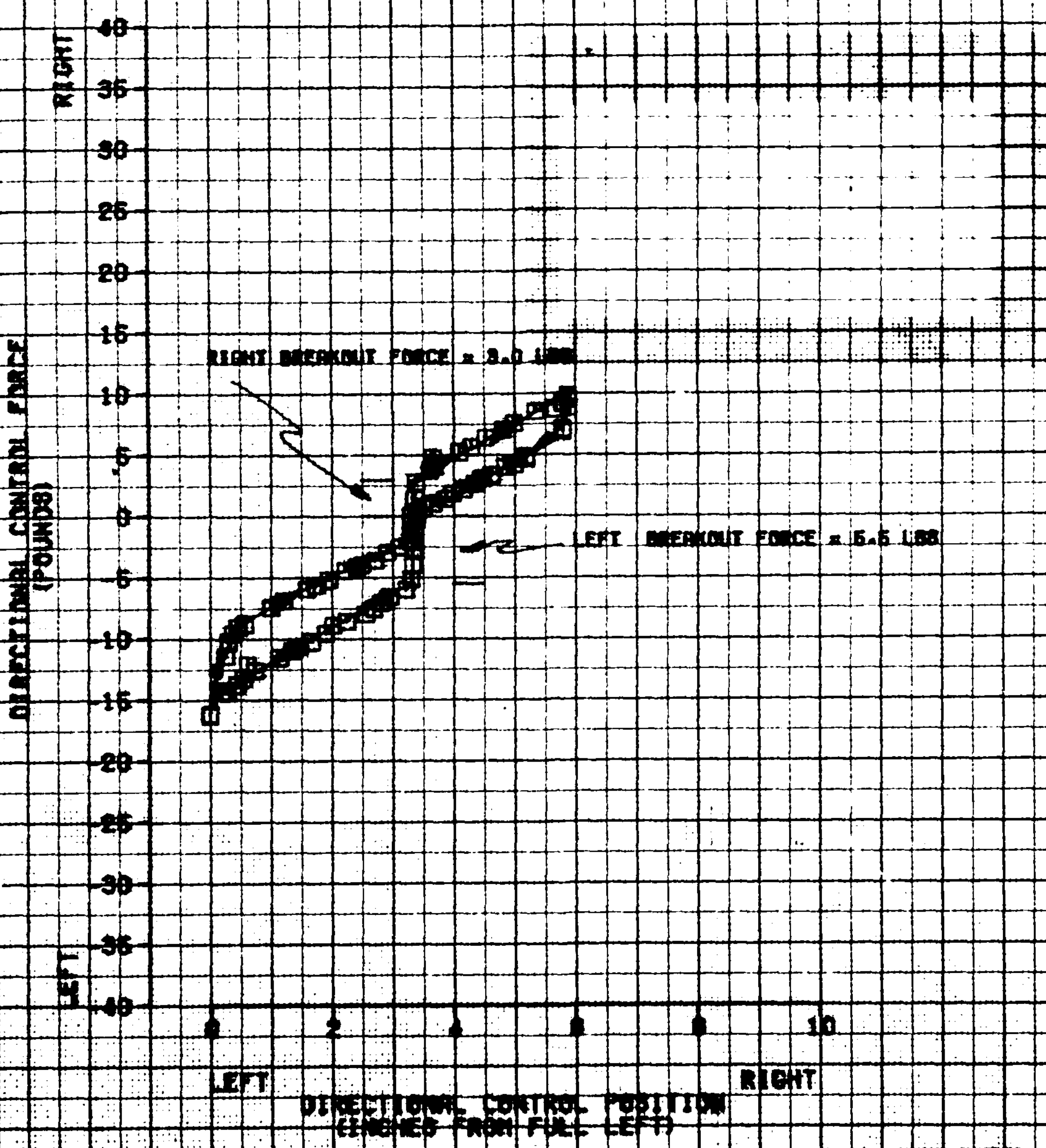


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FIGURE 43
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SAS CONDITION
□	14200.	209.8 (MID)	-.1 (LT)	8630.	26.8	260.	LEVEL FLT	ON

NOTES:
1. CLEAN CONFIGURATION
2. FORCE FEEL ON
3. FORCE TRIM ON

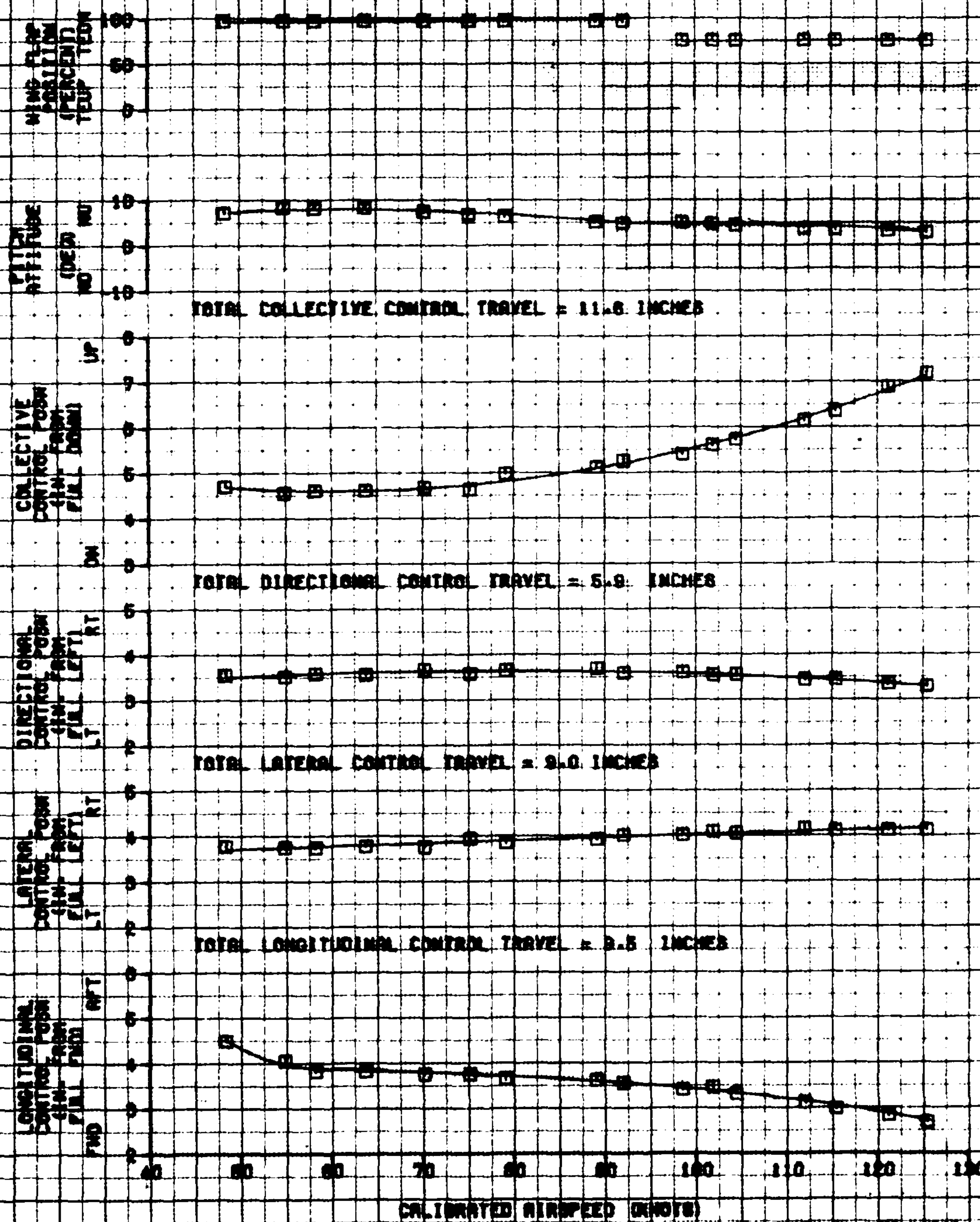

FIGURE 44

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

YAH-64 USA S/N 74-22248

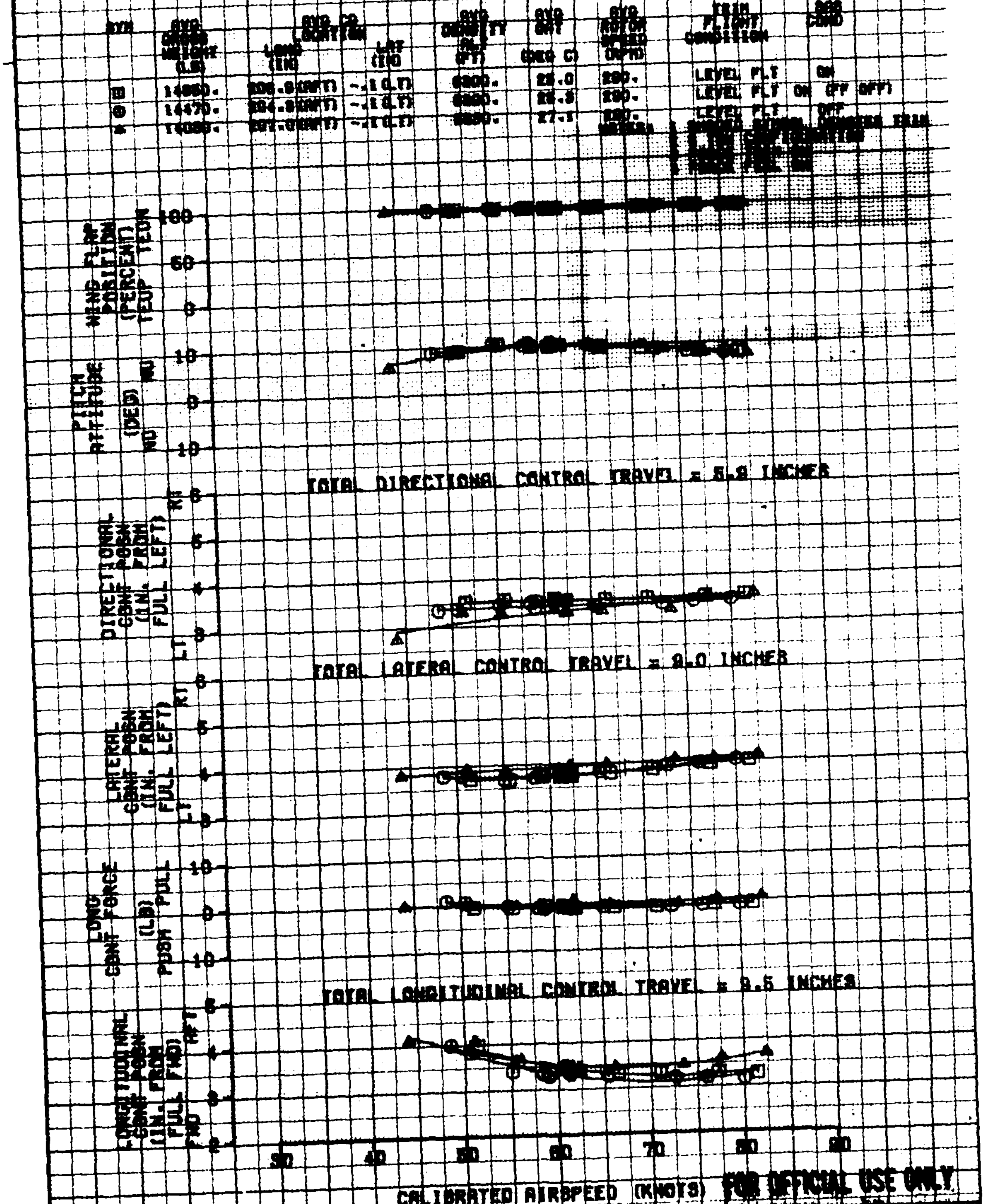

FIGURE 48

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SAS COND
⊡	14600.	206.9 (AFT)	-.1 (LT)	4400.	32.3	290.	LEVEL FLT	ON
⊙	14240.	202.9 (AFT)	-.1 (LT)	6580.	26.3	290.	LEVEL FLT	OFF

NOTES: 1 SHADED SYMBOL DENOTES TRIM
2 S TOW CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON

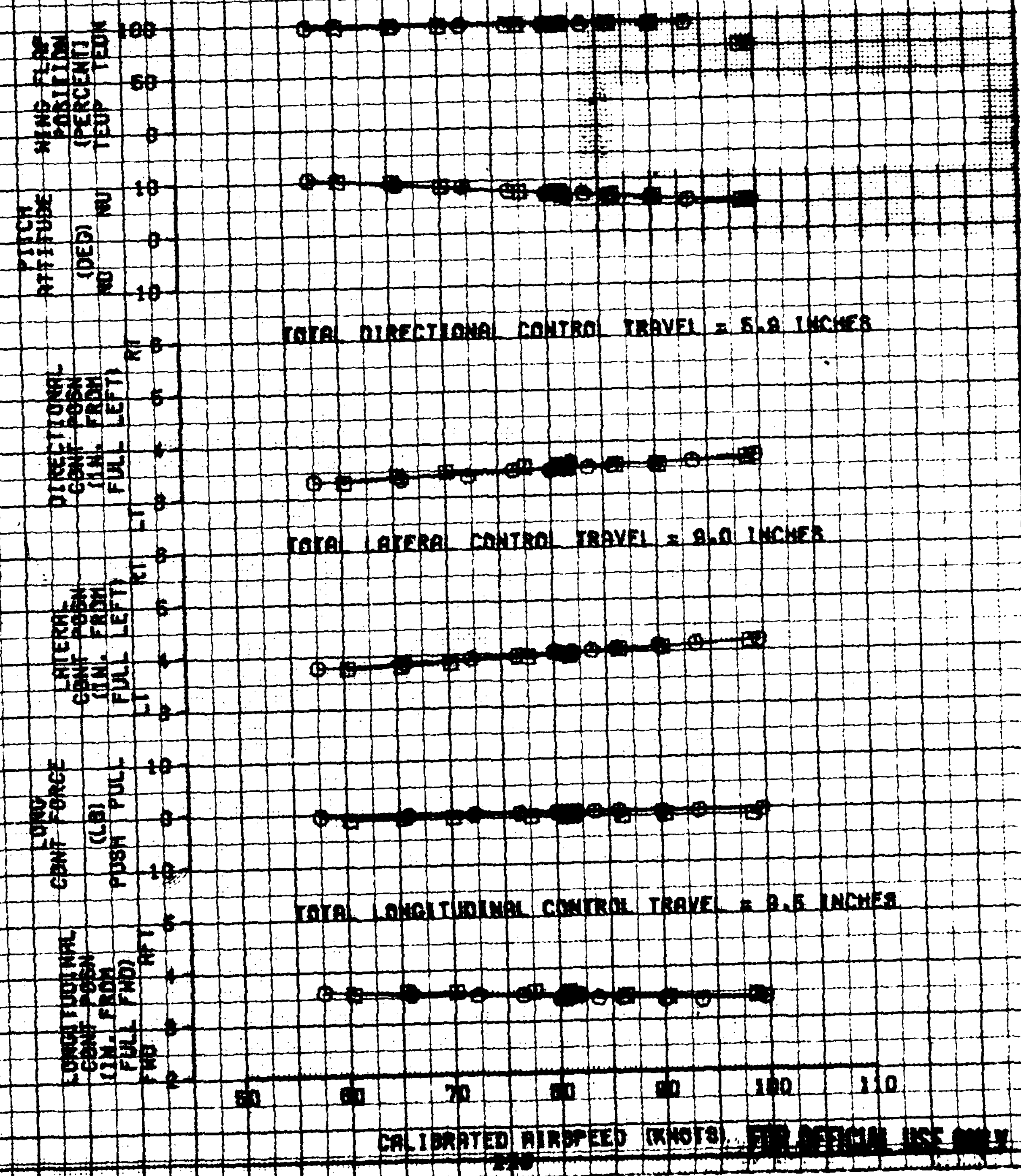

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FIGURE 46

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DAS COND
□	14600.	200.9(AFT)	-.1 (LT)	5780.	26.6	290.	LEVEL FLT	ON
○	14810.	200.9(AFT)	-.1 (LT)	6980.	27.2	290.	LEVEL FLT	OFF

NOTES:
1 SHADED SYMBOL DENOTES TRI
2 8 TOW CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON

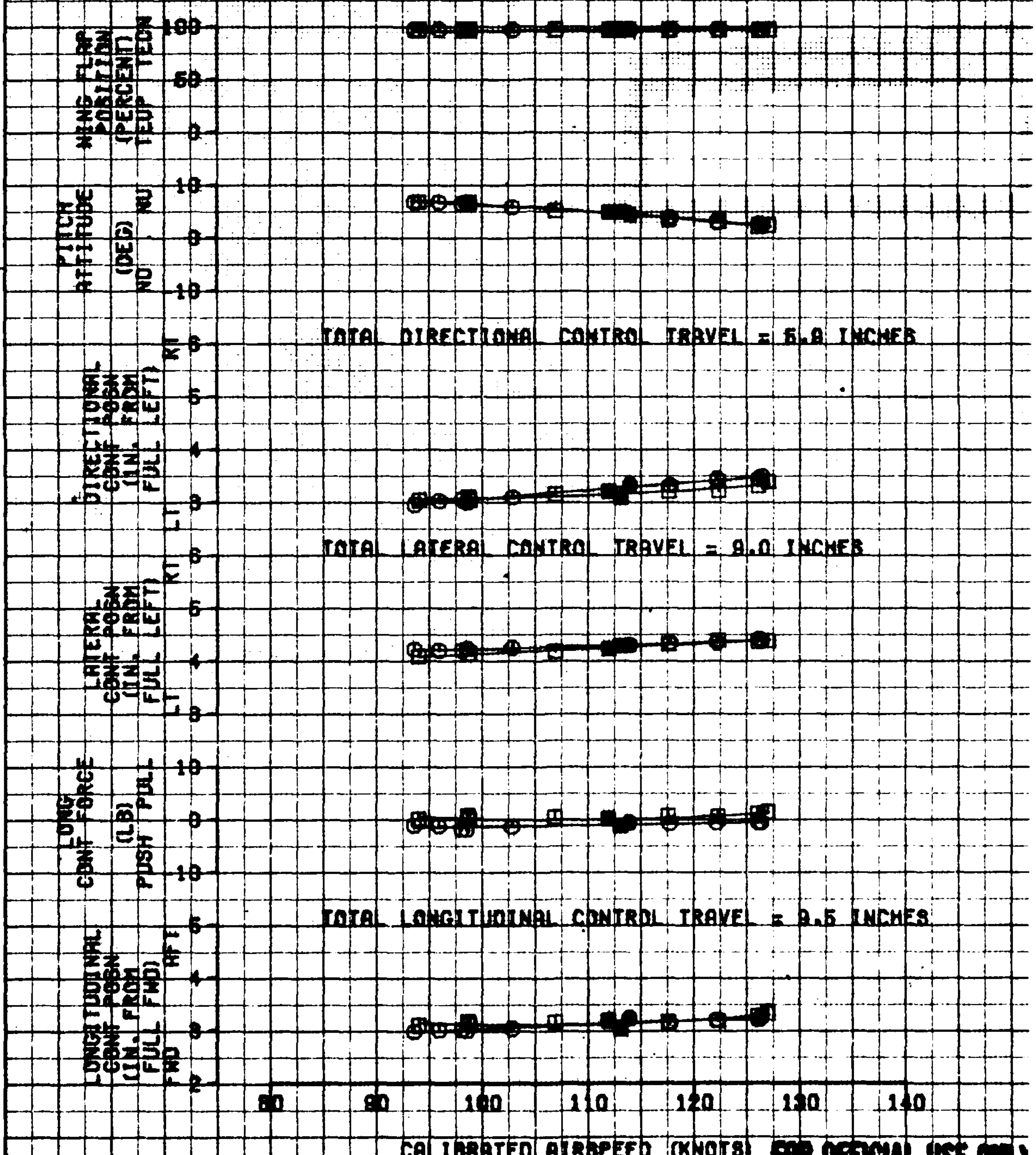


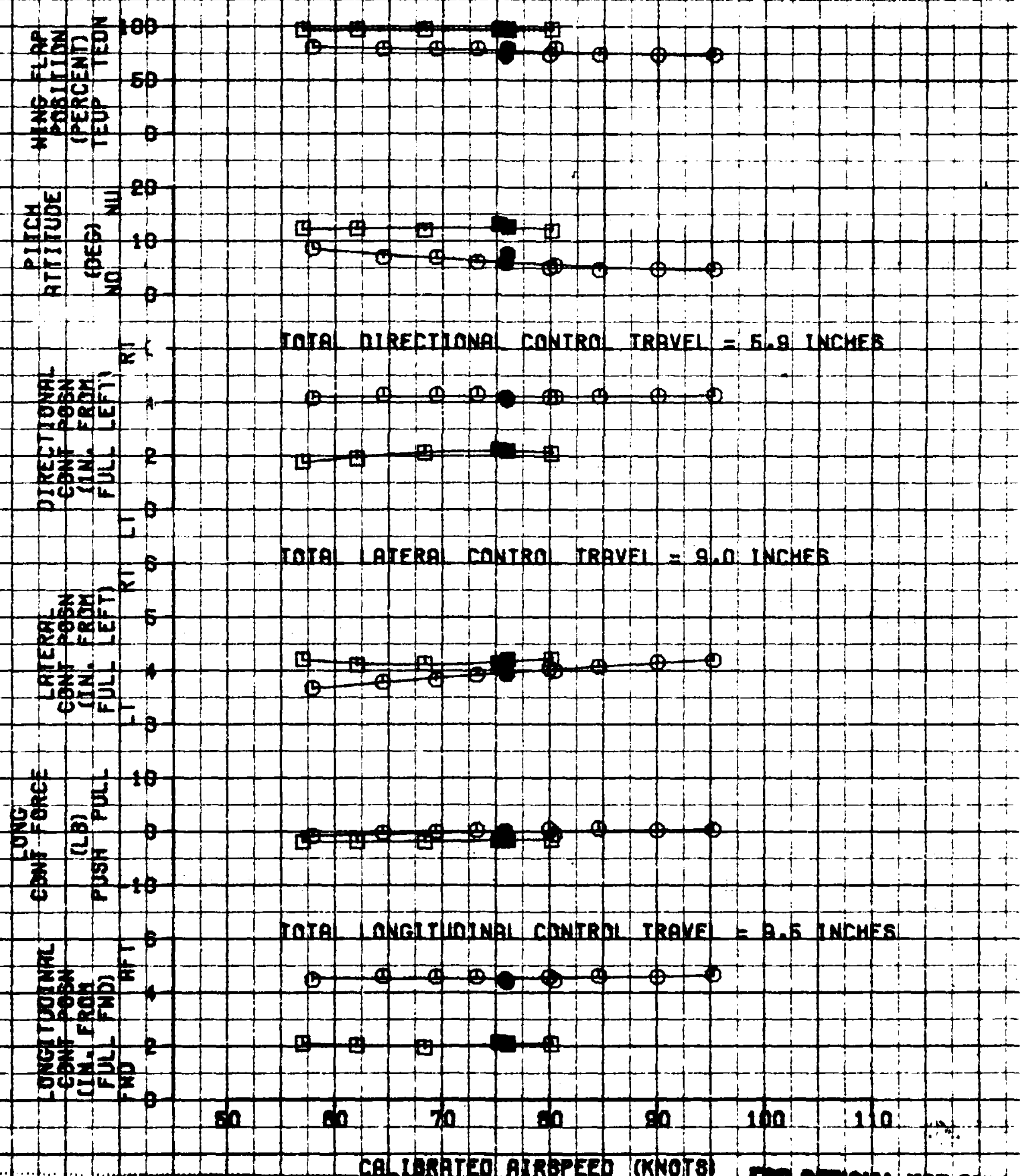
FIGURE 47

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SAS COND
□	14440.	206.8(AFT)	-.1(LT)	6800.	22.7	288.	CLIMB	ON
○	14480.	206.8(AFT)	-.1(LT)	6590.	22.0	290.	AUTO	ON

NOTES: 1 SHADED SYMBOL DENOTES TRIM
2 8 TOW CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON



STATIC LATERAL-DIRECTIONAL STABILITY

YAH-64 UDA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS CONDITION
□	14600.	207.0 (AFT)	-.1 (LT)	5400.	24.0	288.	62.	LEVEL FLIGHT	ON
○	14560.	207.0 (AFT)	-.1 (LT)	5520.	22.5	288.	60.	LEVEL FLIGHT	OFF

NOTES: 1 SHADED SYMBOL DENOTES TRIM
2 3 TOW CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON

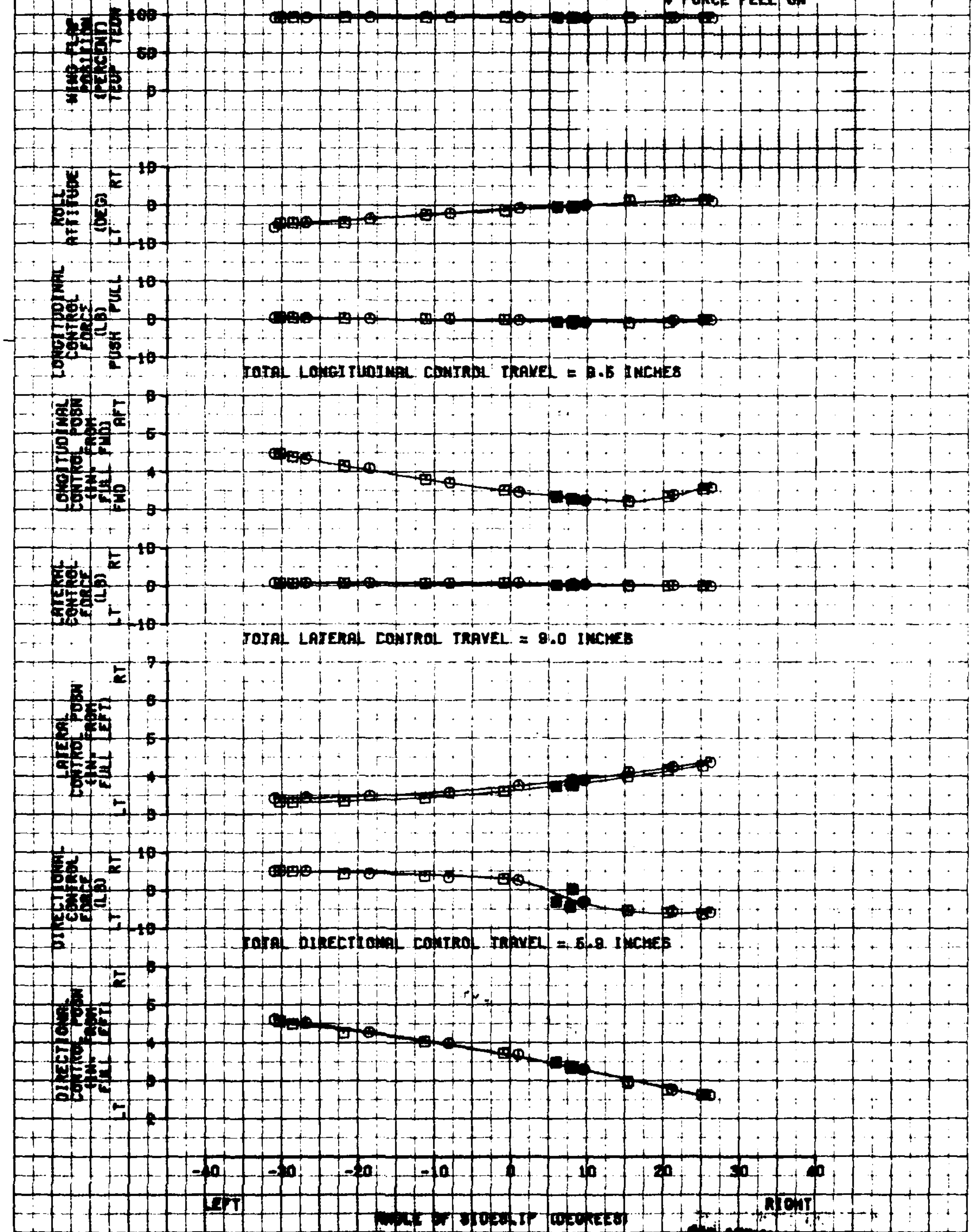


FIGURE 49

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-64 USA S/N 74-22249

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS CONDITION
○	14840.	207.0 (AFT)	-.1 (LT)	4000.	26.0	290.	80.	LEVEL FLIGHT	ON
□	14430.	207.0 (AFT)	-.1 (LT)	5200.	27.0	290.	82.	LEVEL FLIGHT	OFF

NOTES: 1 SHADED SYMBOL DENOTES TRIM
2 8 TON CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON

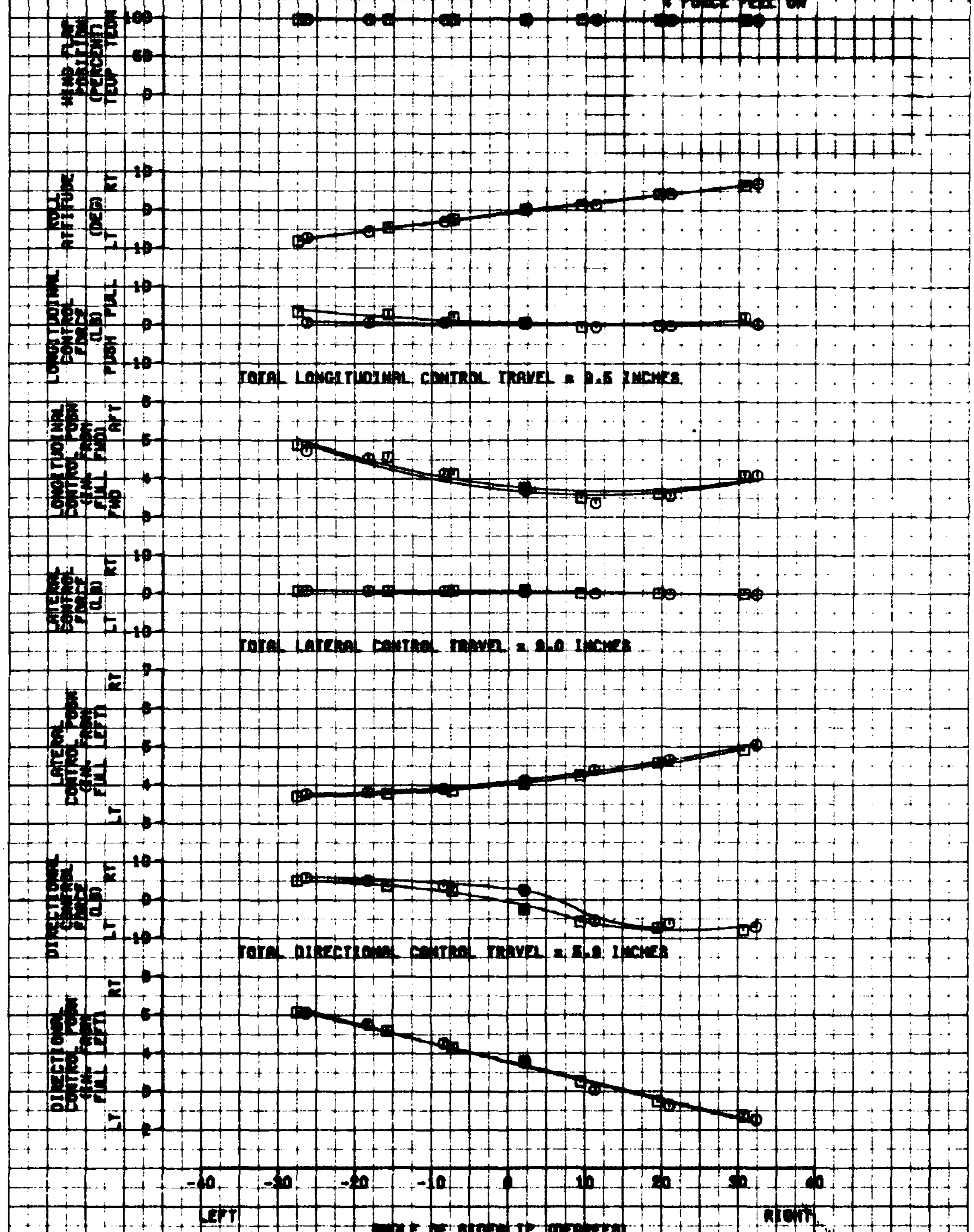


FIGURE 39

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-64 USA S/N 74-22249

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS CONDITION
□	14140.	207.1 (AFT)	-.1 (LT)	6420.	23.5	288.	113.	LEVEL FLIGHT	ON
○	14180.	207.1 (AFT)	-.1 (LT)	5860.	30.0	280.	112.	LEVEL FLIGHT	OFF

NOTES: 1 SHADED SYMBOL DENOTES TRIM
2 8 TOW CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON

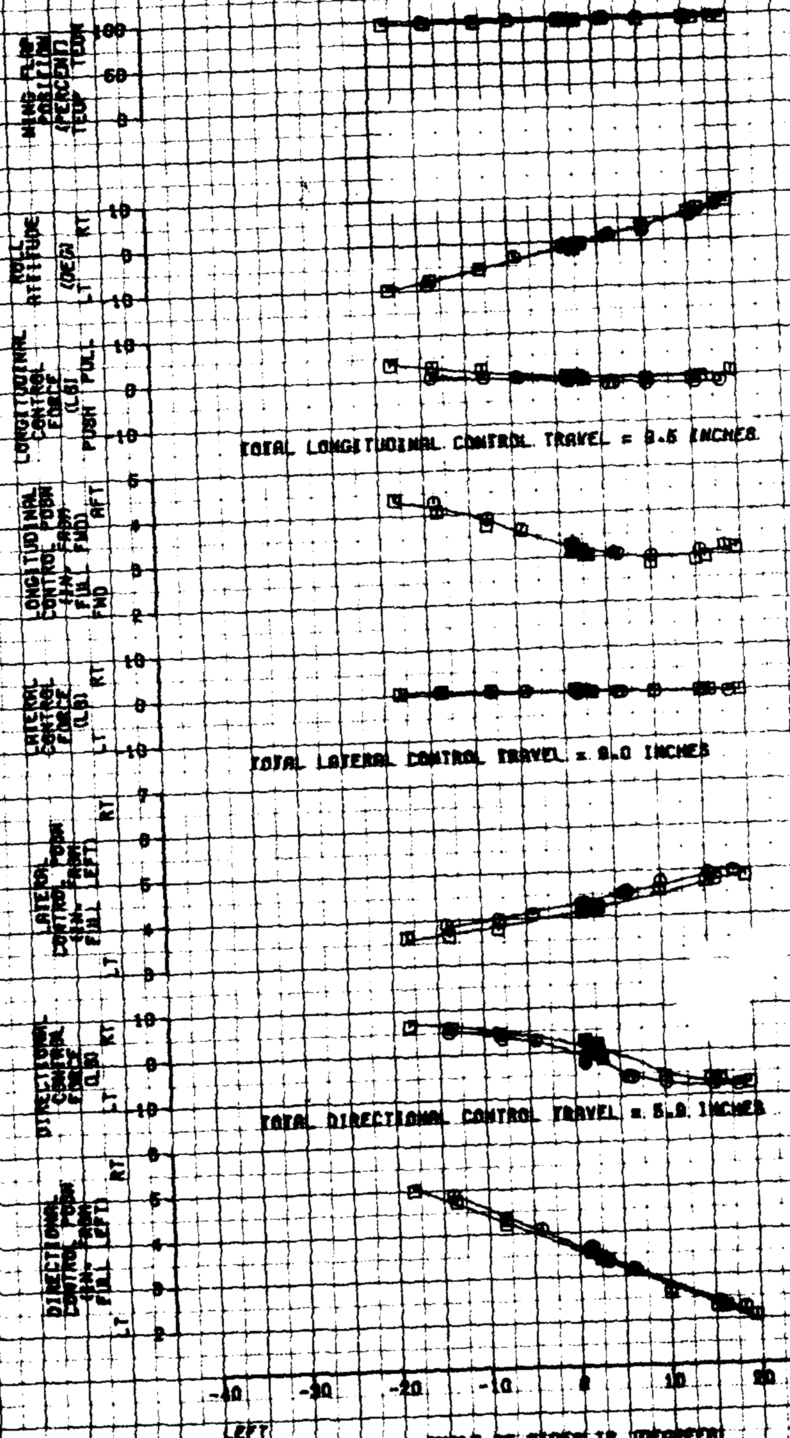


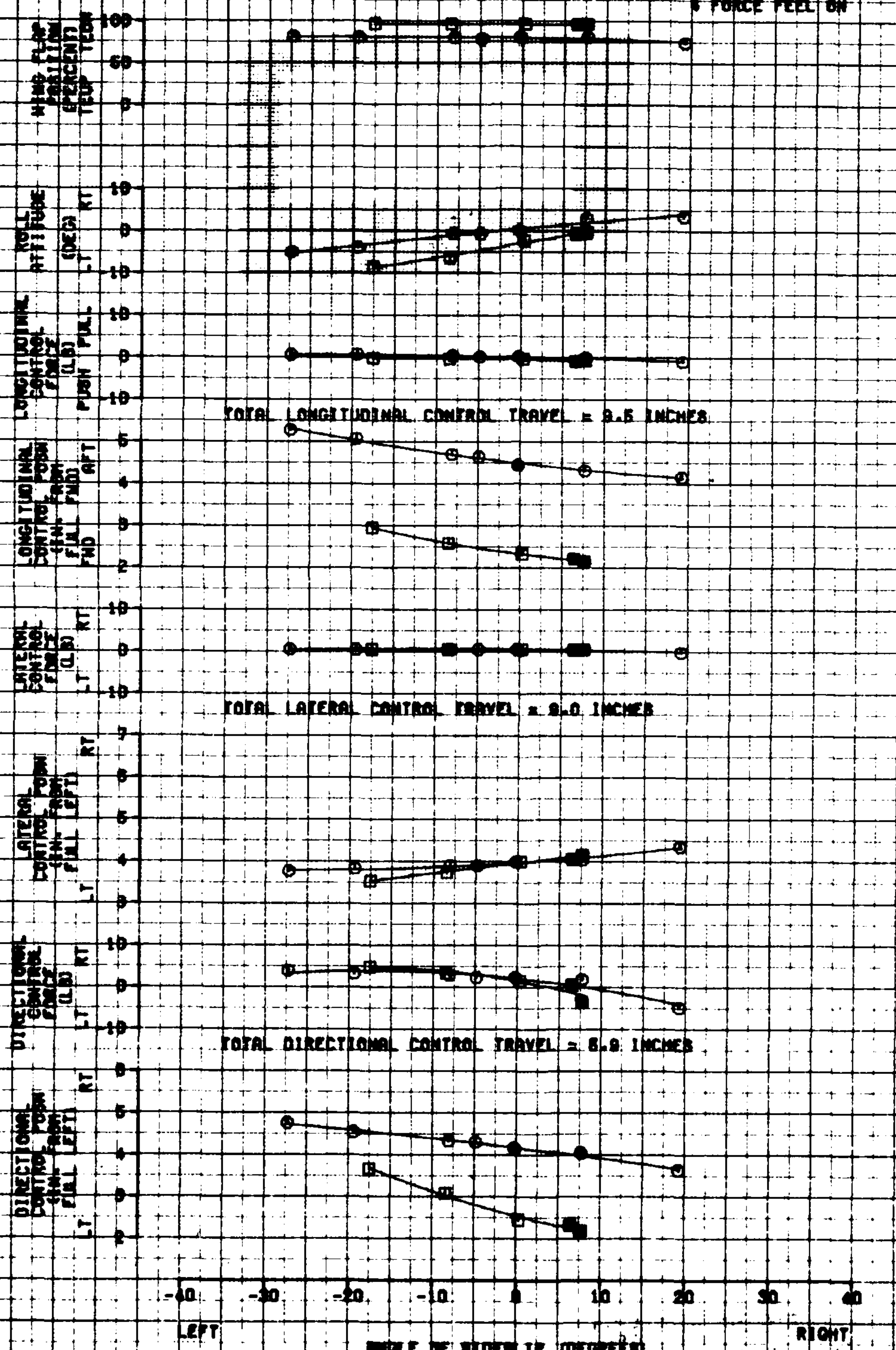
FIGURE 81

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS CONDITION
□	14170.	206.9 (AFT)	-.1 (LT)	6600.	23.5	290.	76.	CLIMB	ON
○	14520.	206.9 (AFT)	-.1 (LT)	6780.	22.5	292.	75.	AUTO	ON

NOTES: 1 SHADED SYMBOL DENOTES TRIM
2 8 TOW CONFIGURATION
3 FORCE TRIM ON
4 FORCE FEEL ON



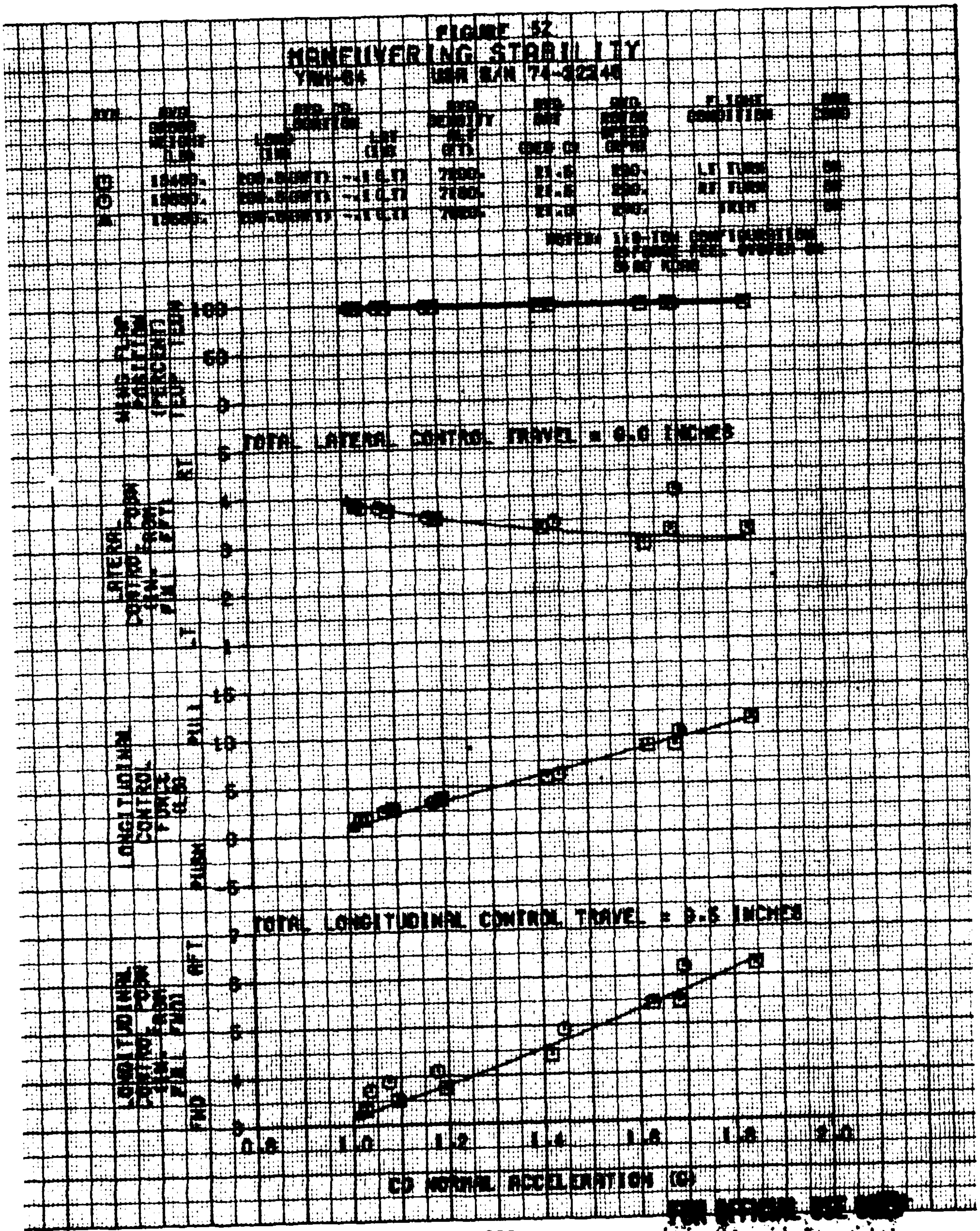


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FIGURE 53

MANEUVERING STABILITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	SAS COND
⊡	13620.	208.8 (AFT)	-.1 (LT)	6640.	23.0	289.	LT TURN	ON
⨀	13670.	208.8 (AFT)	-.1 (LT)	6420.	23.5	290.	RT TURN	ON
▲	13780.	208.8 (AFT)	-.1 (LT)	7000.	22.5	289.	TRIM	ON

NOTES: 1) 8-TOW CONFIGURATION
2) FORCE FEEL SYSTEM ON
3) 88 KCAS

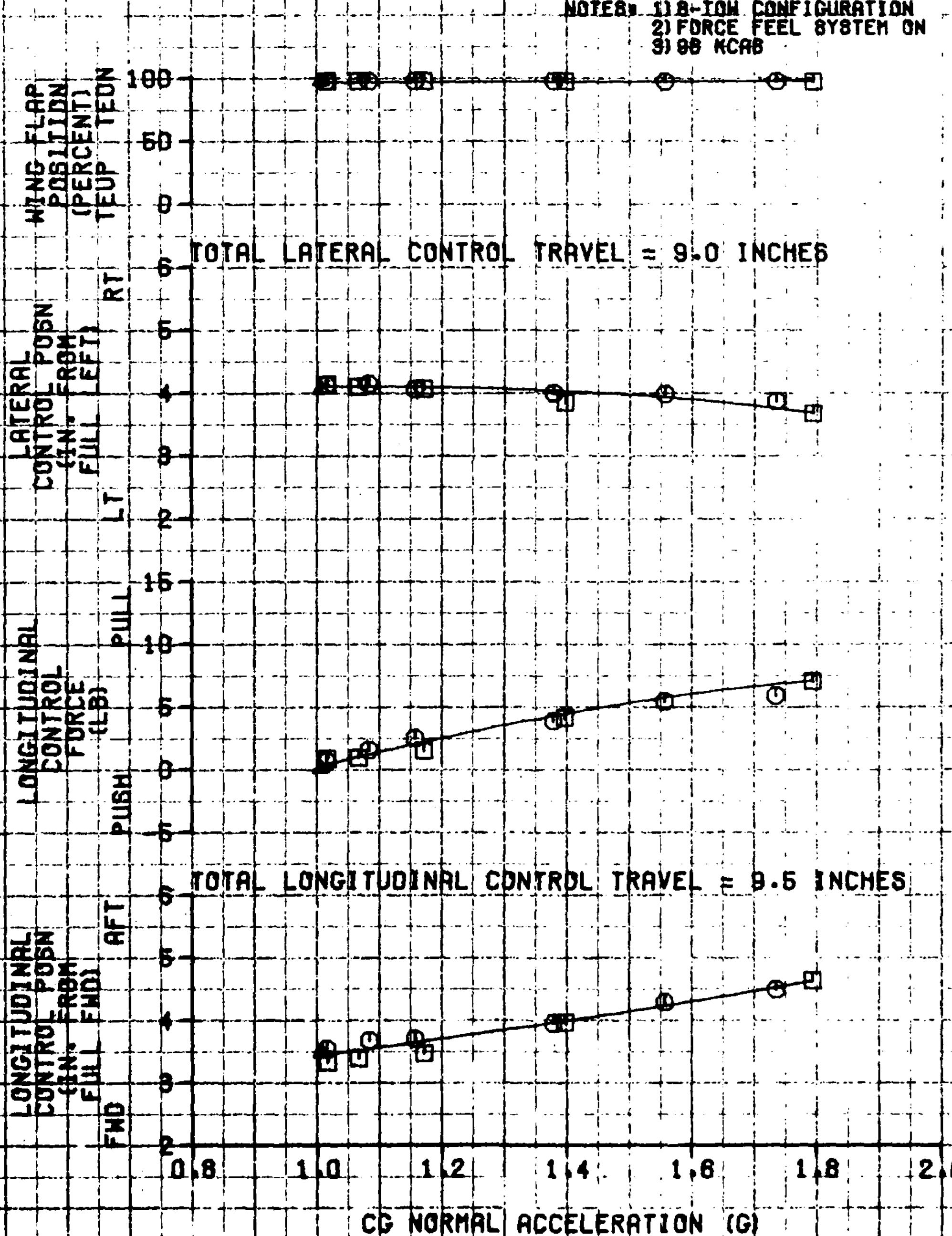

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FIGURE 54

MANEUVERING STABILITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	SAS COND
□	14130.	207.0(AFT)	-.1(LT)	7000.	30.0	289.	LT TURN	ON
○	14000.	207.0(AFT)	-.1(LT)	6900.	32.0	289.	RT TURN	ON
▲	14200.	207.0(AFT)	-.1(LT)	7140.	31.0	289.	TRIM	ON

NOTES: 1) 8-TOW CONFIGURATION
2) FORCE FEEL SYSTEM ON
3) [illegible]

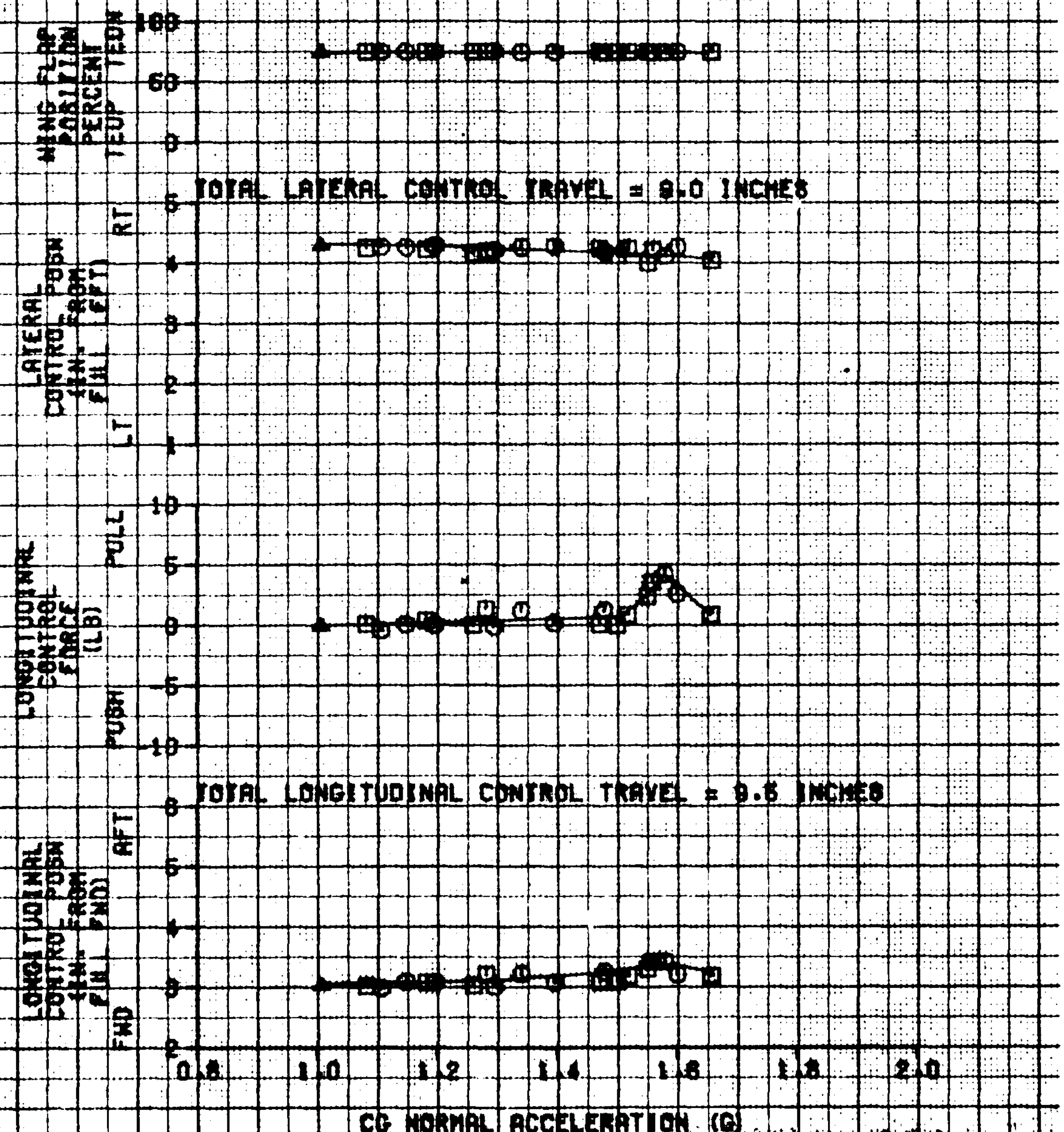


FIGURE 55

MANEUVERING STABILITY
SYMMETRICAL PUSHOVERS AND PULLUPS
YAH-64 USA S/N 74-22248

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	SAS CONDITION
13970.	208.8 (AFT)	-.1 (LT)	7480.	21.0	289.	ON

NOTES: 1) 8-TOW CONFIGURATION
2) FORCE FEEL SYSTEM ON
3) STICK MOVEMENT AT 80 KCAS

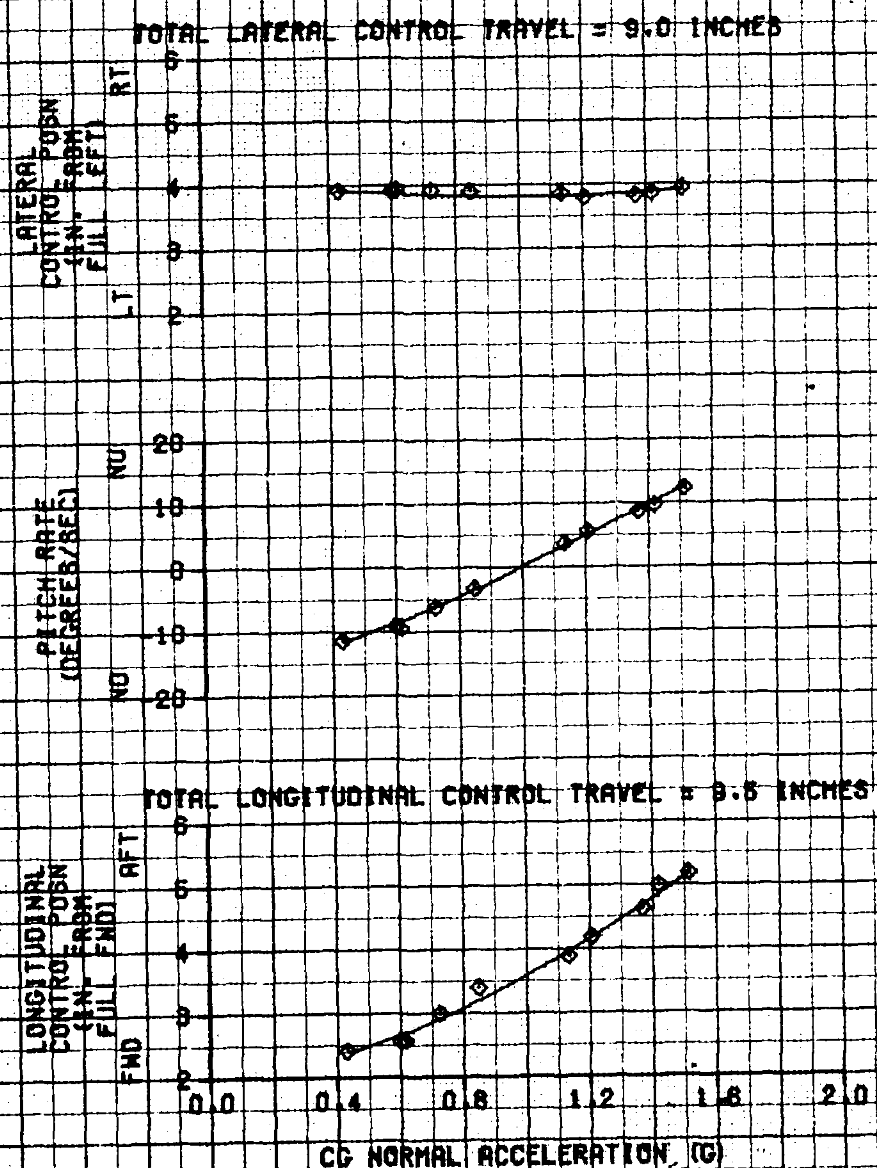


FIGURE 56

MANEUVERING STABILITY
SYMMETRICAL PUSHOVERS AND PULLUPS
YAH-64 USA S/N 74-22248

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	SAS CONDITION
15870.	206.9(AFT)	-.5 (LT)	6640.	25.0	289.	ON

NOTES: 1) 8-TOW CONFIGURATION
2) FORCE FEEL SYSTEM ON
3) STICK MOVEMENT AT 97 KCAS

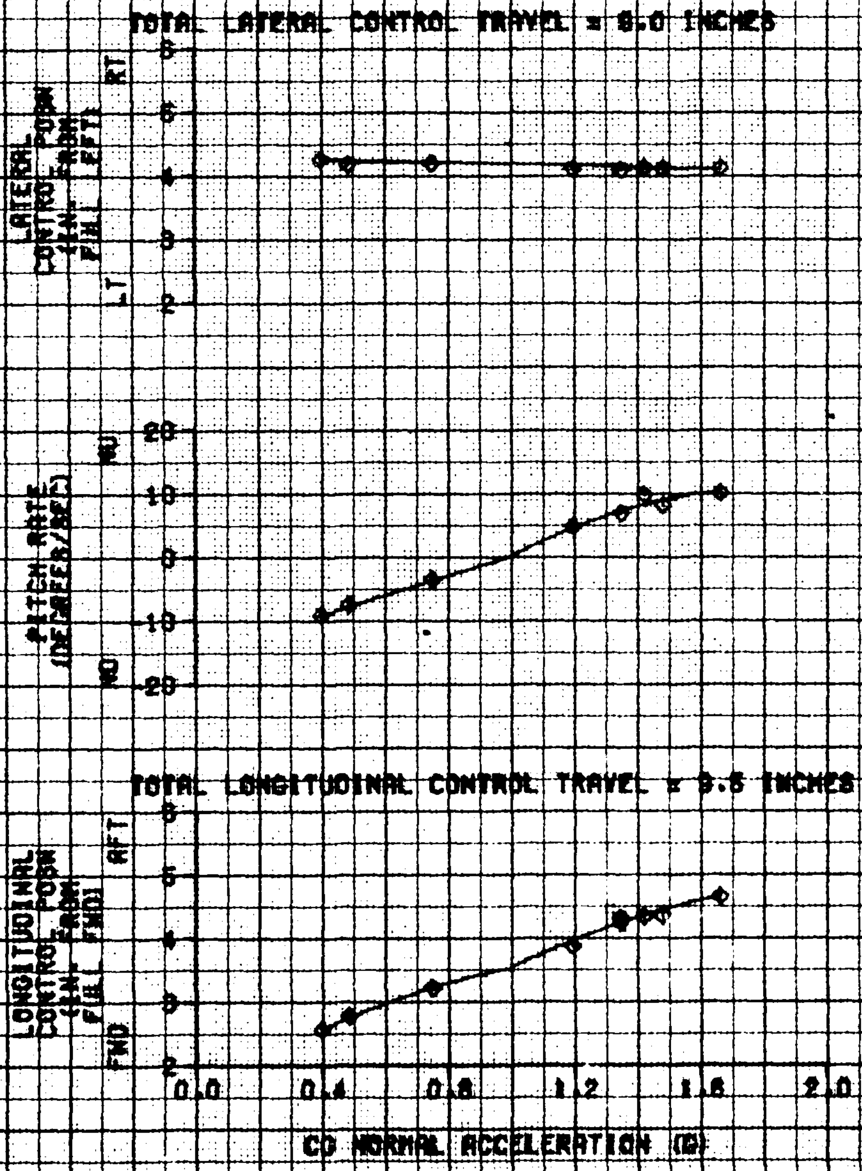
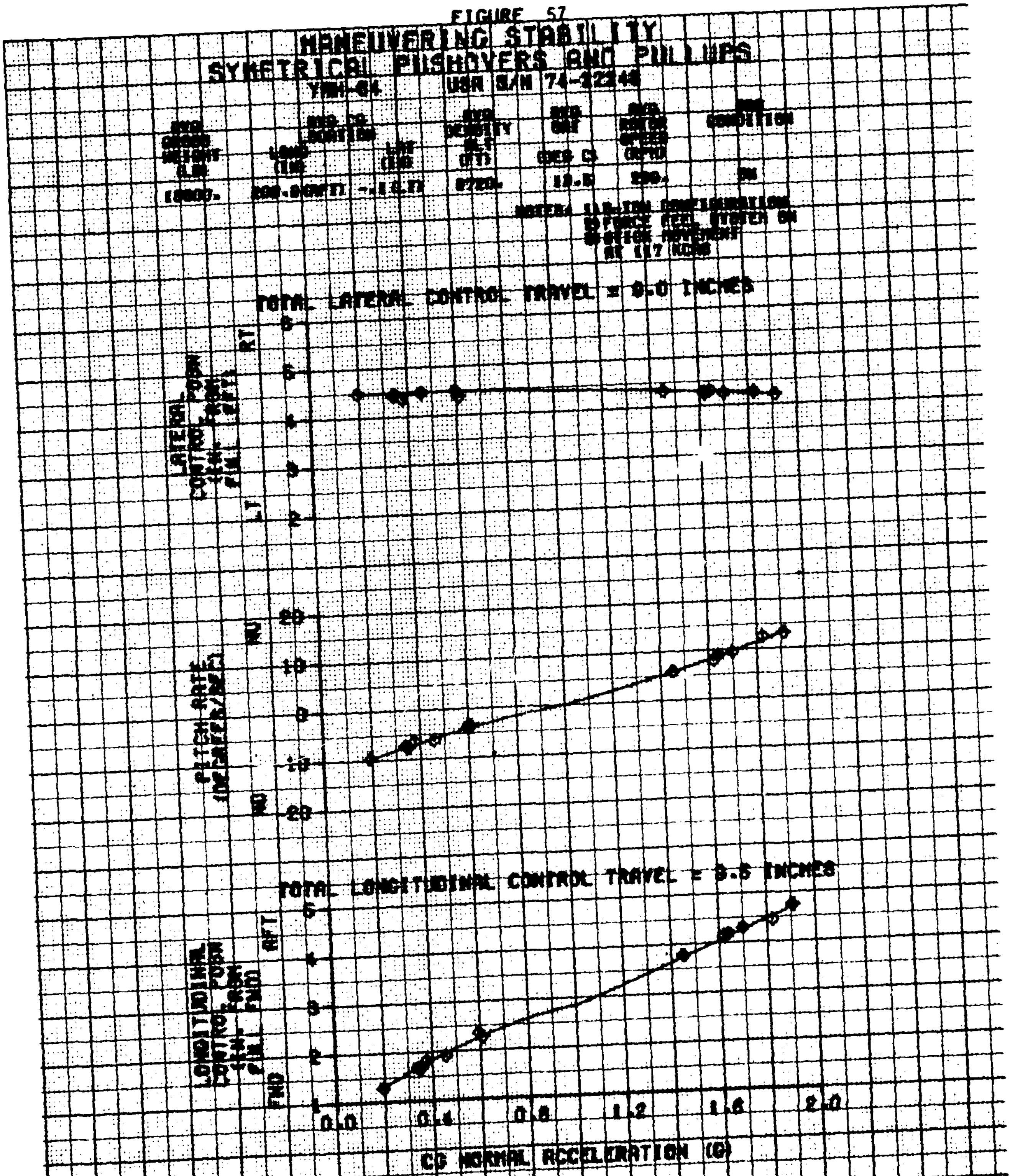


FIGURE 57



MANEUVERING STABILITY
SYMETRICAL PUSHOVERS AND PULLUPS
YAH-64 USA S/N 74-22248
CONDITION
TOTAL LATERAL CONTROL TRAVEL = 9.0 INCHES
LATERAL CONTROL POSITION
RT
LT
PITCH RATE
NU
ND
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.5 INCHES
LONGITUDINAL CONTROL POSITION
AFT
FWD
0.0
0.4
0.8
1.2
1.6
2.0
CG NORMAL ACCELERATION (G)

FIGURE 58
AFT LONGITUDINAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14630.	207.0 (AFT)	-.1 (LT)	5800.	26.5	289.	92.	ON

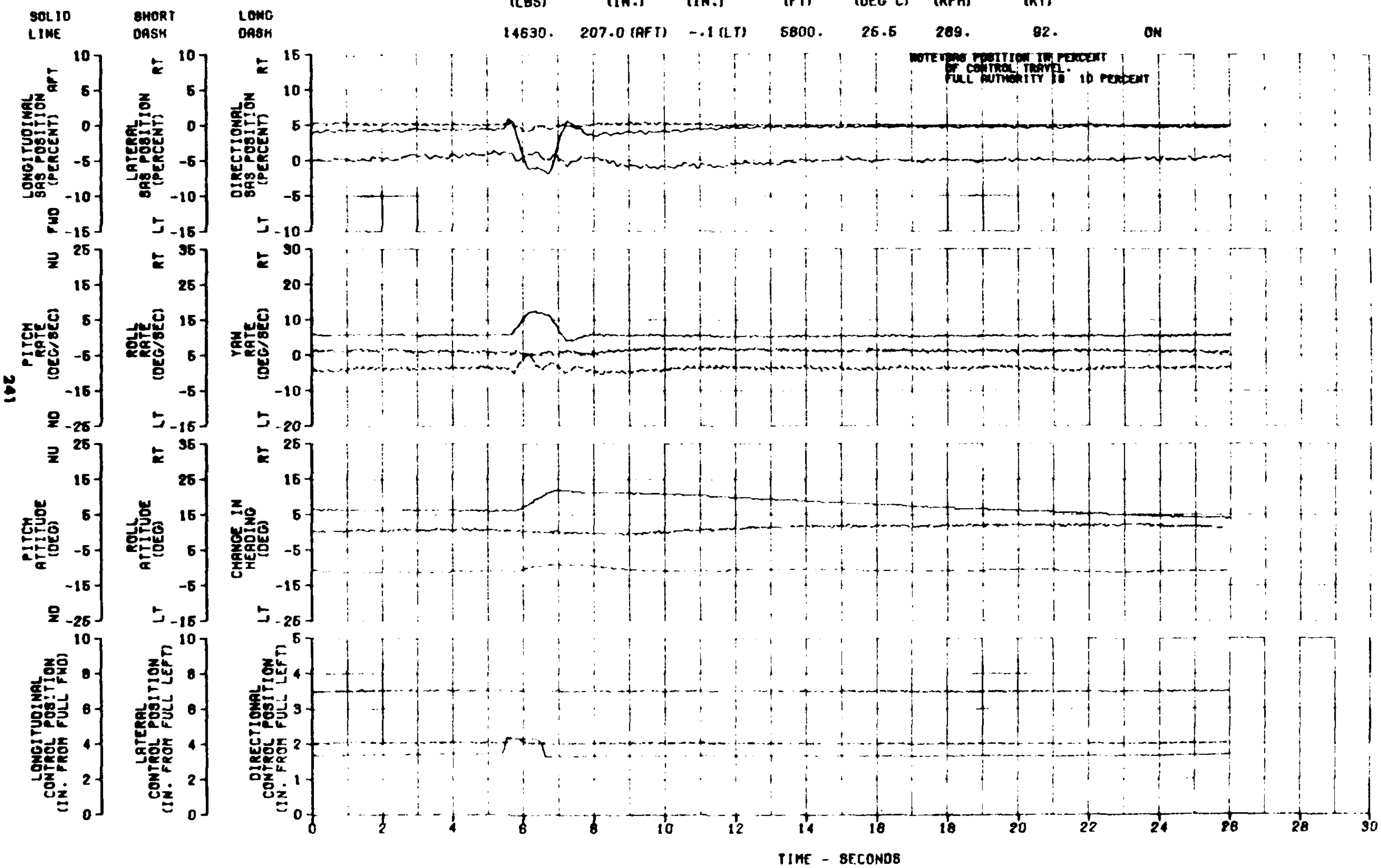


FIGURE 59
AFT LONGITUDINAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

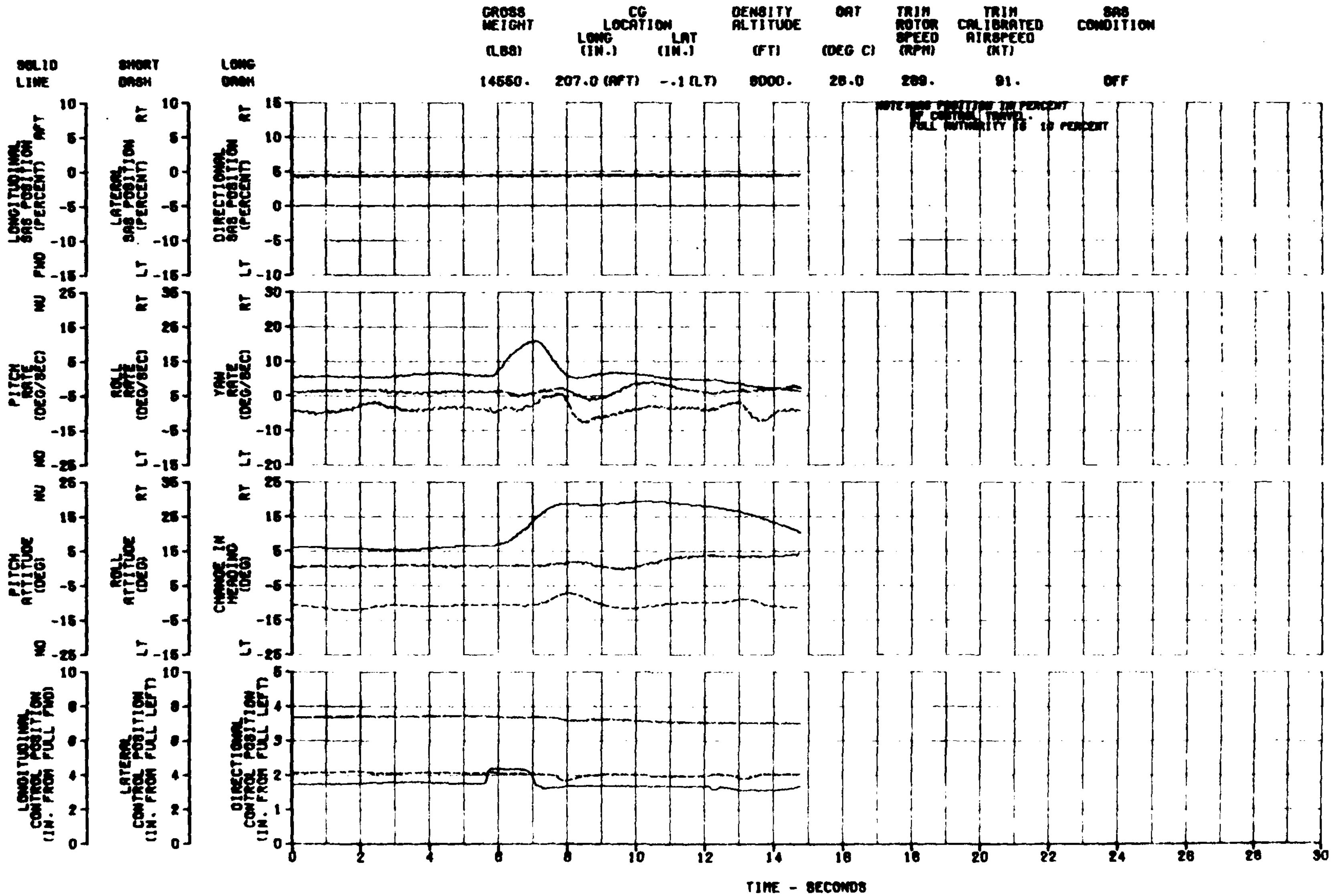


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FIGURE 60
FORWARD LONGITUDINAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14580.	208.9 (AFT)	-.1 (LT)	5720.	14.5	290.	108.	ON

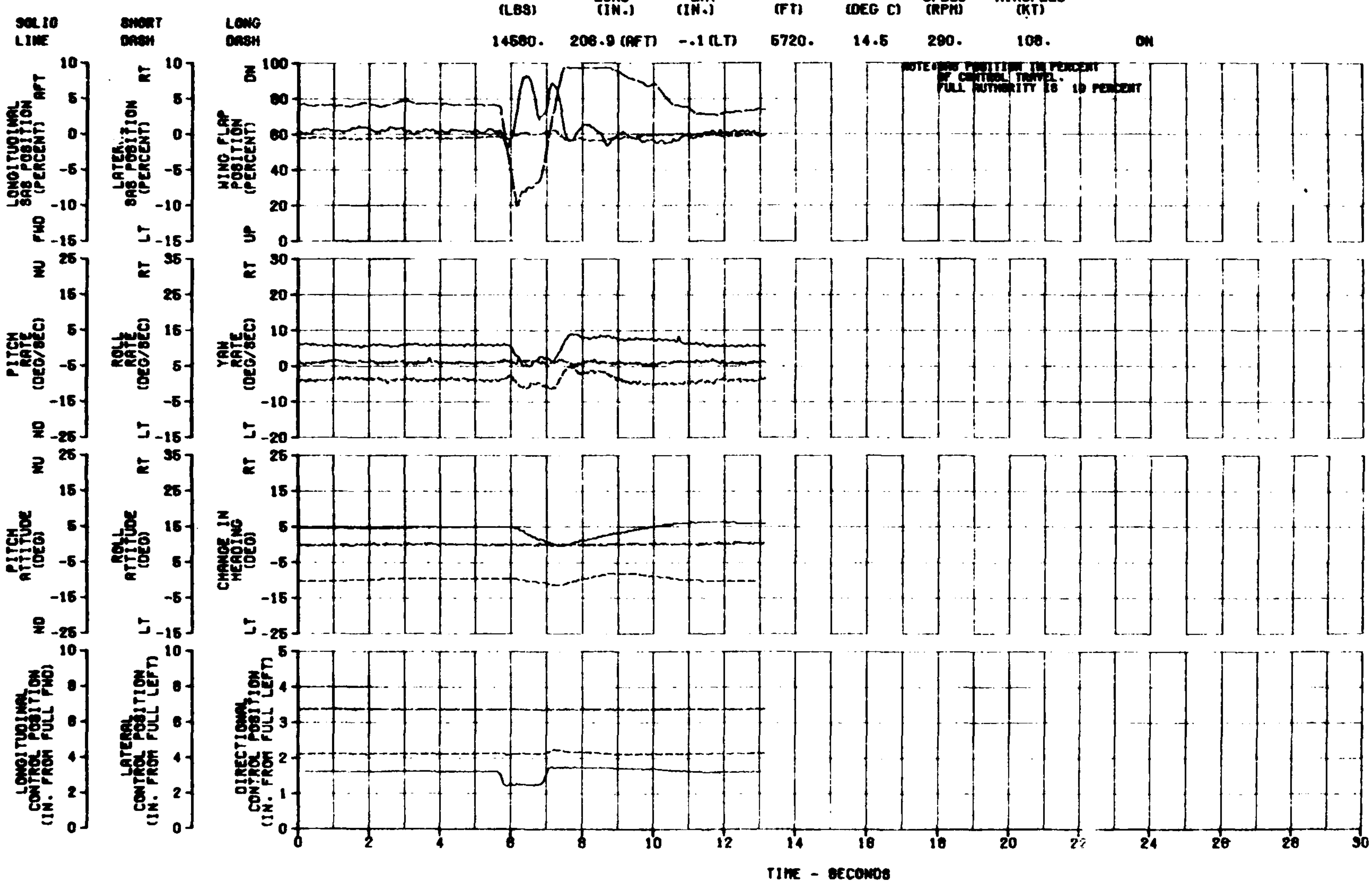

FIGURE 61

AFT LONGITUDINAL SHORT TERM RESPONSE

YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14580.	206.9 (AFT)	-.1 (LT)	5660.	14.5	291.	107.	ON

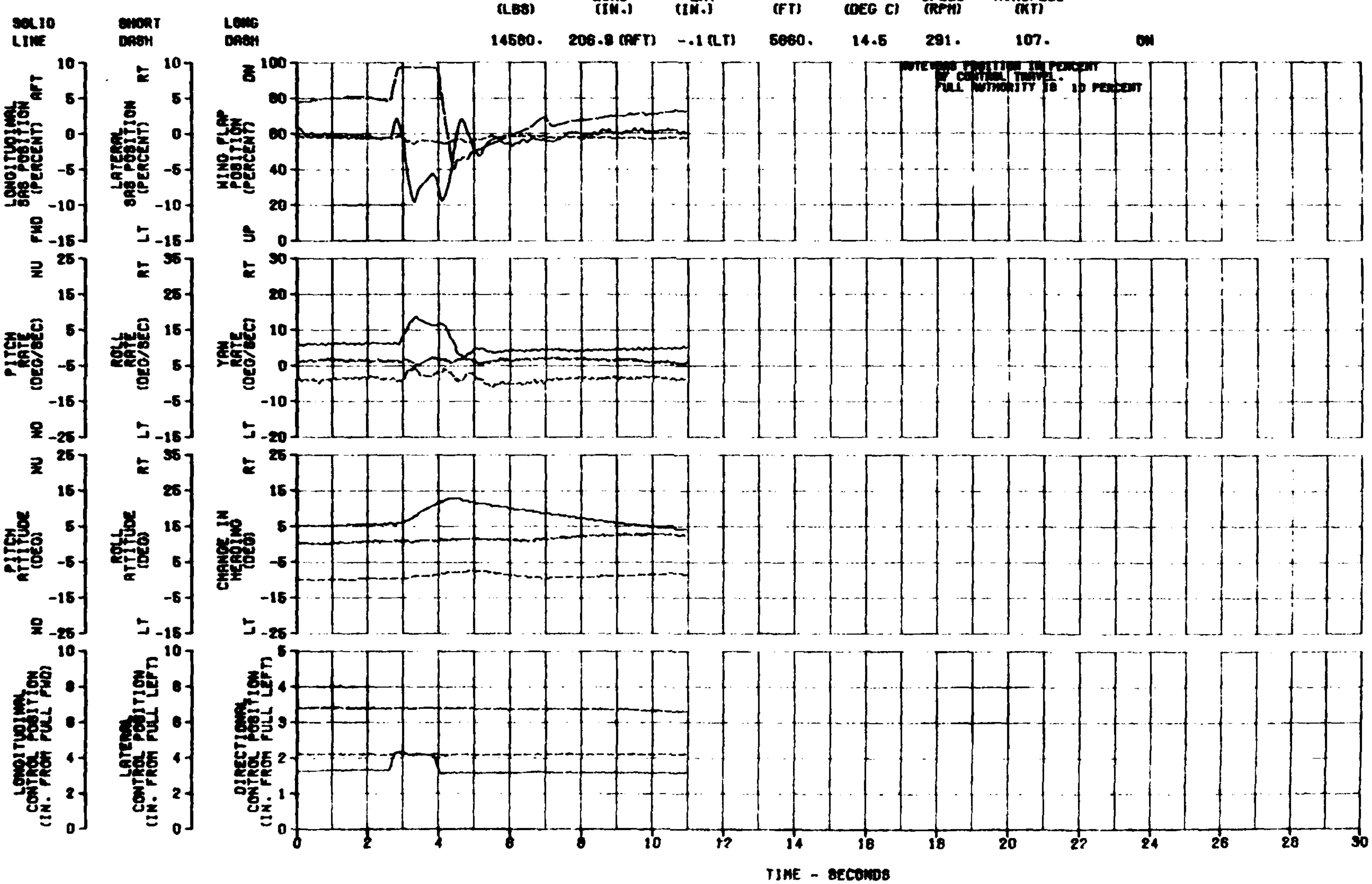

FIGURE 62

LONGITUDINAL LONG TERM RESPONSE

YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14350.	206.9 (AFT)	-.1 (LT)	5800.	14.5	289.	106.	ON

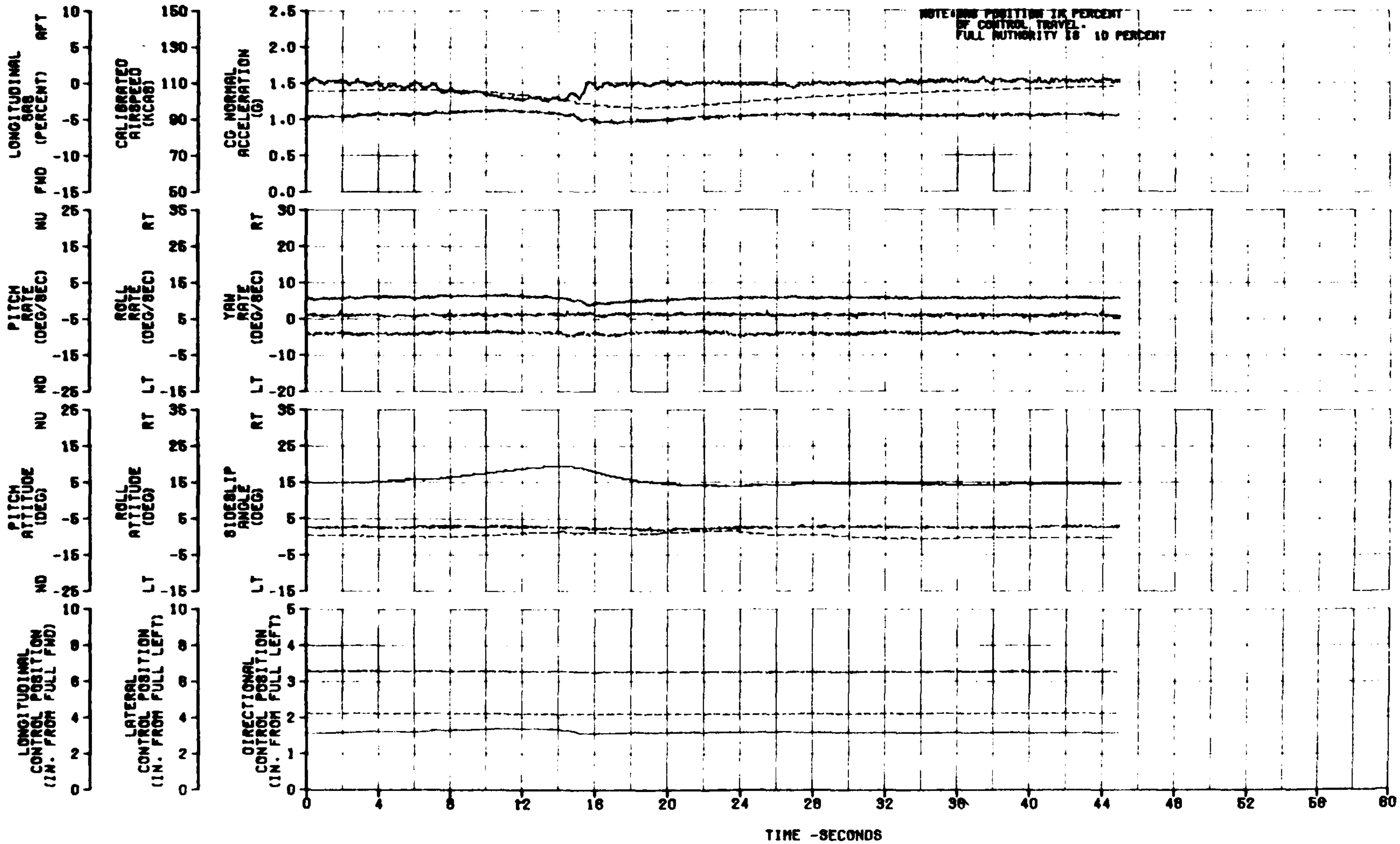

FIGURE 83
LONGITUDINAL LONG TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14240.	207.0 (AFT)	-.1 (LT)	5580.	15.0	291.	106.	OFF

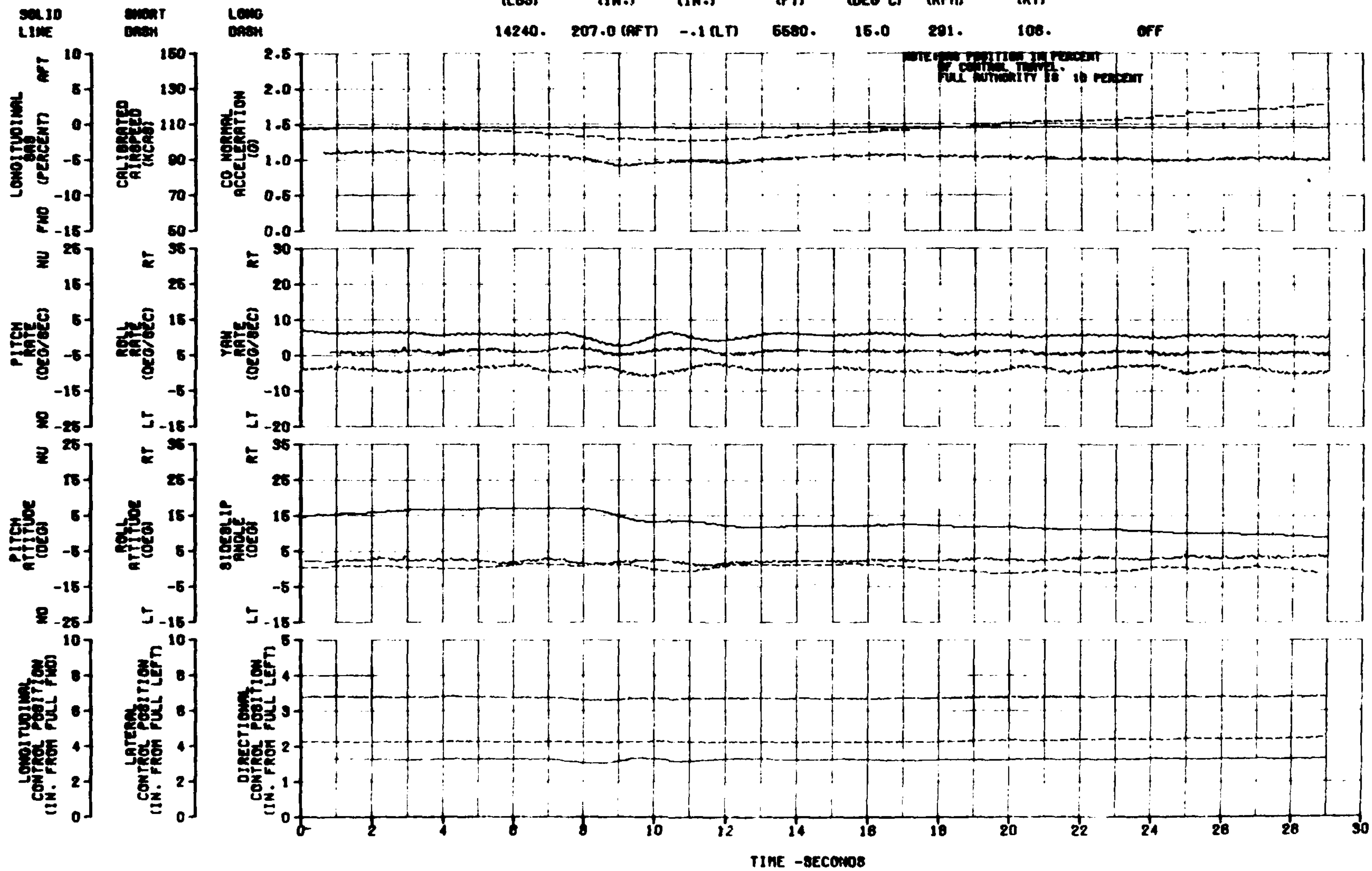

FIGURE 64

HORIZONTAL STABILIZER DYNAMIC CHARACTERISTICS

YAH-64 USA S/N 74-22248

GROSS WEIGHT	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14520.	207.0 (AFT)	-.1 (LT)	4740.	21.0	290.	53.	ON

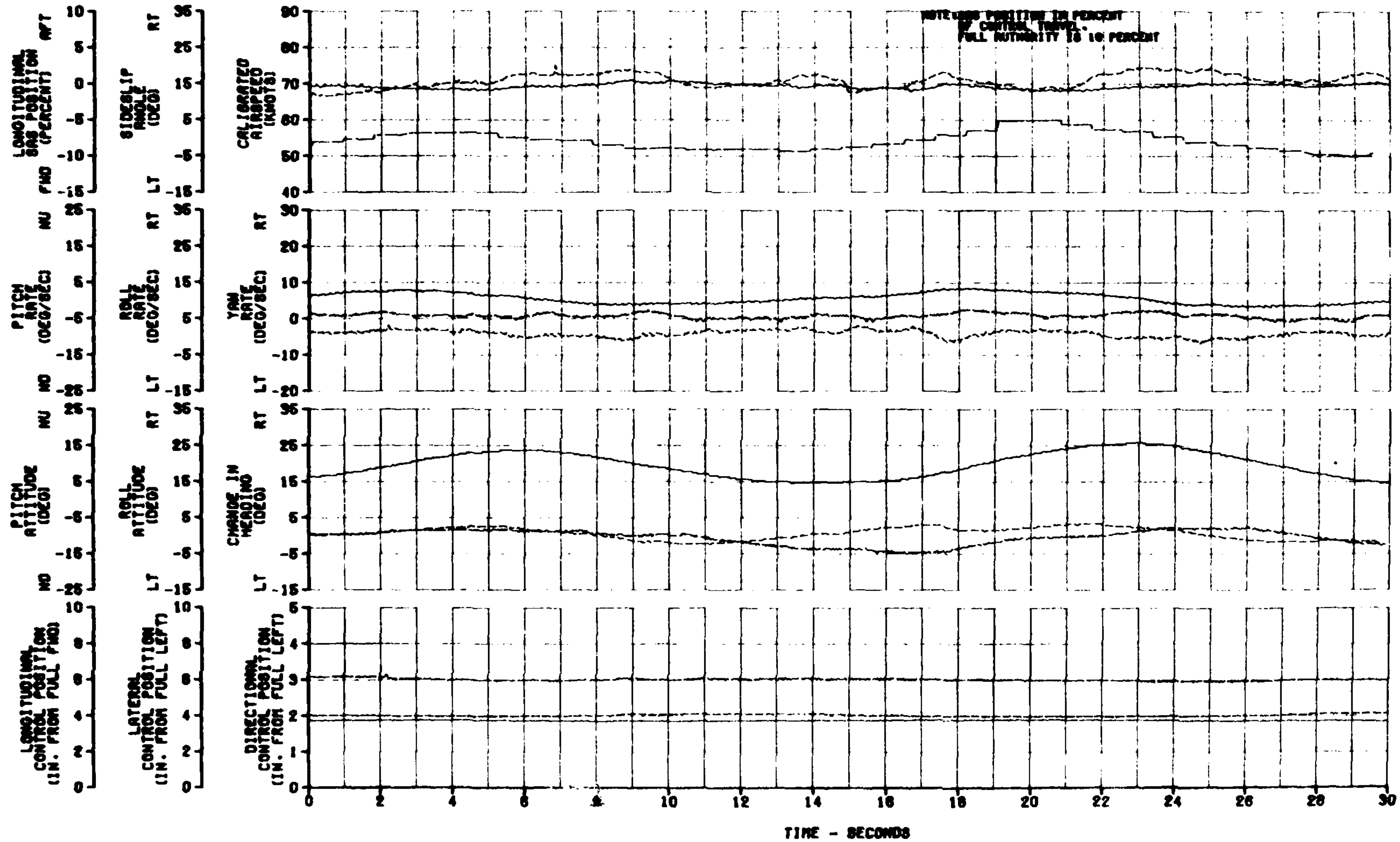

FIGURE 65
LEFT LATERAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14330.	207.1 (AFT)	-.1 (LT)	5980.	26.0	288.	91.	ON

SOLID LINE	SHORT DASH	LONG DASH
LONGITUDINAL SAS POSITION (PERCENT)	LATERAL SAS POSITION (PERCENT)	DIRECTIONAL SAS POSITION (PERCENT)
PITCH RATE (DEG/SEC)	ROLL RATE (DEG/SEC)	YAW RATE (DEG/SEC)
PITCH ATTITUDE (DEG)	ROLL ATTITUDE (DEG)	CHANGE IN HEADING (DEG)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)	LATERAL CONTROL POSITION (IN. FROM FULL LEFT)	DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)

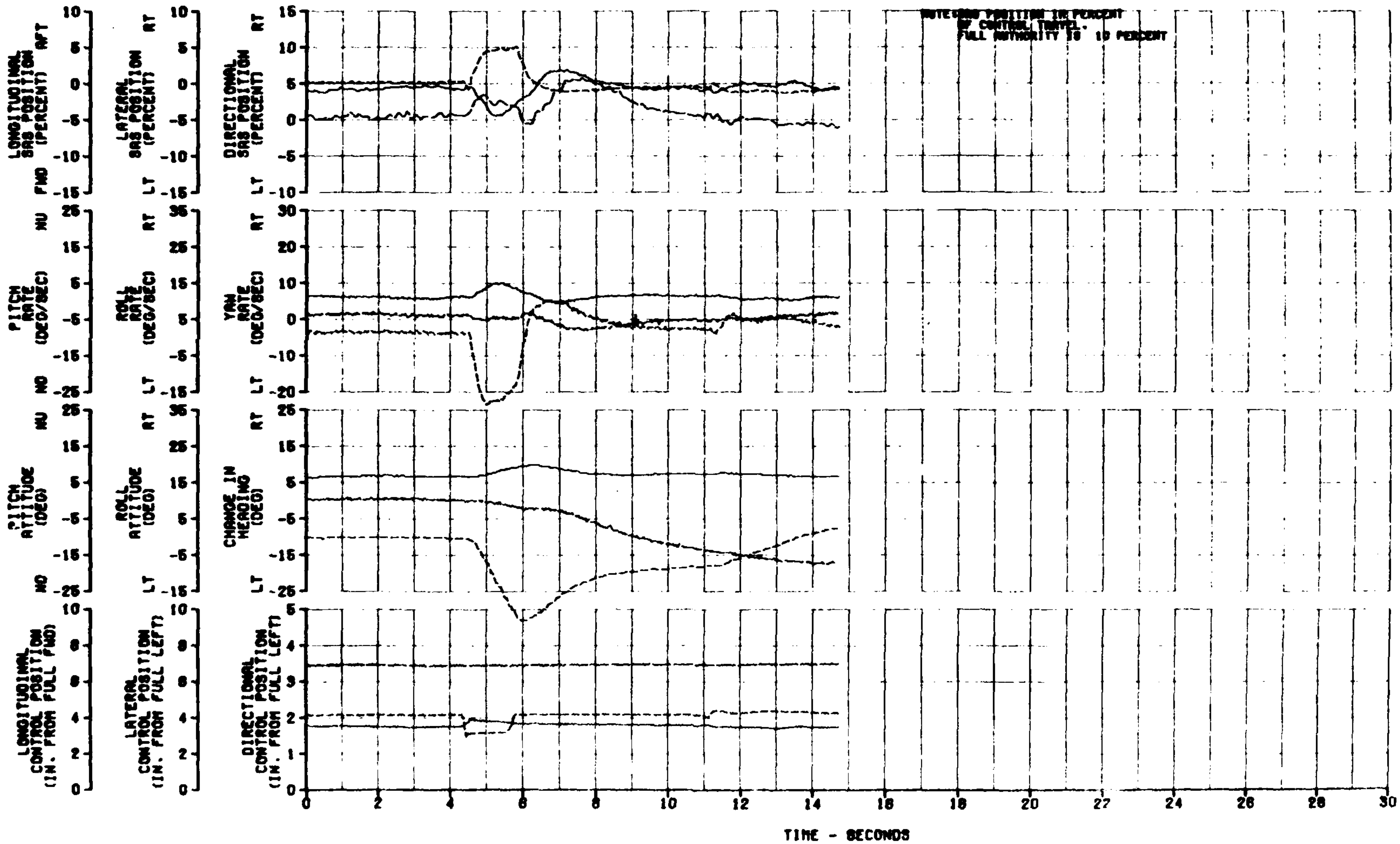

FIGURE 86

LEFT LATERAL SHORT TERM RESPONSE

YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14270.	207.1 (AFT)	-.1 (LT)	6040.	25.0	289.	92.	OFF

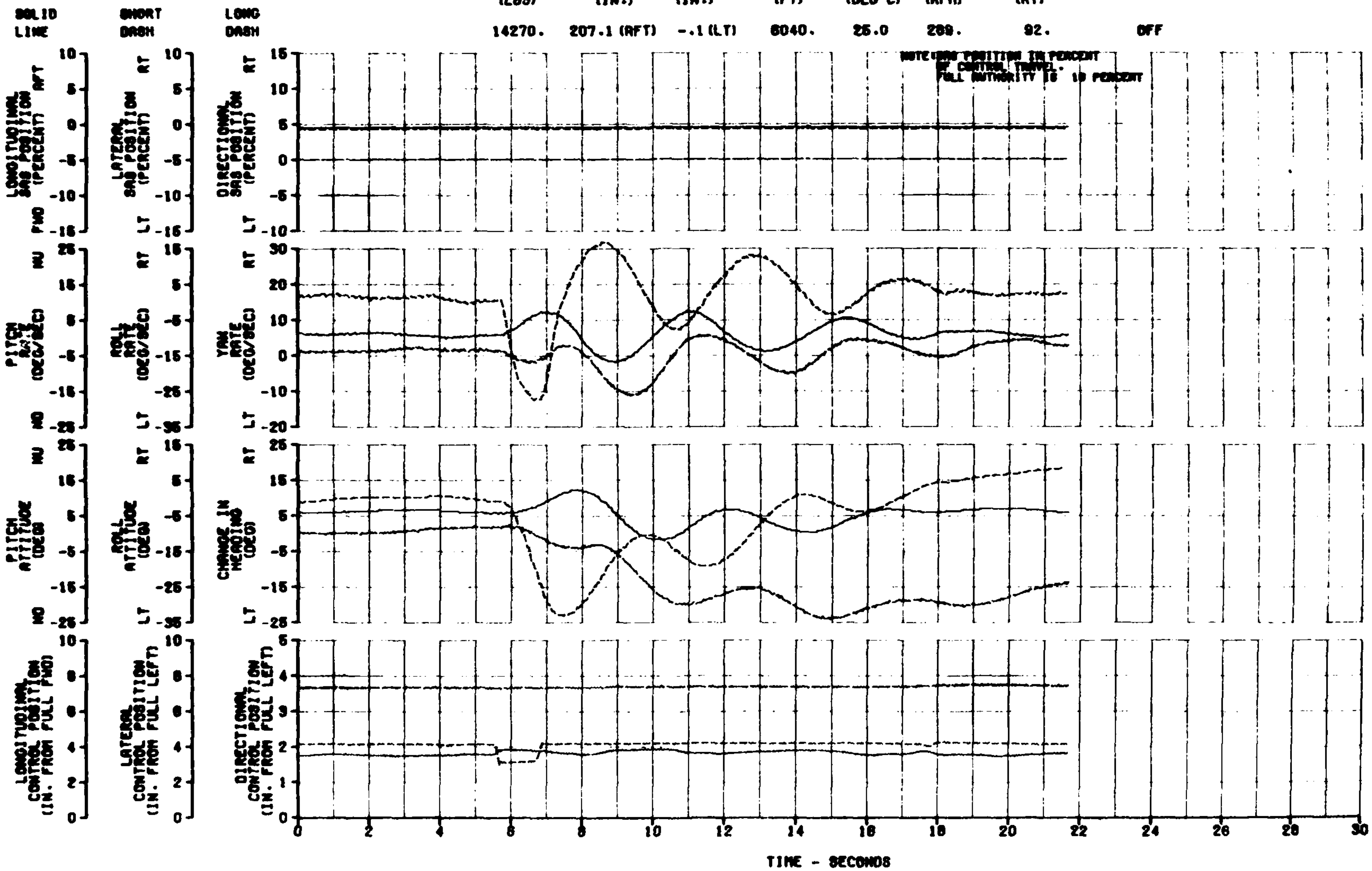

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FIGURE 67
RIGHT LATERAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14230.	207.1 (AFT)	-.1 (LT)	8280.	25.0	288.	89.	ON

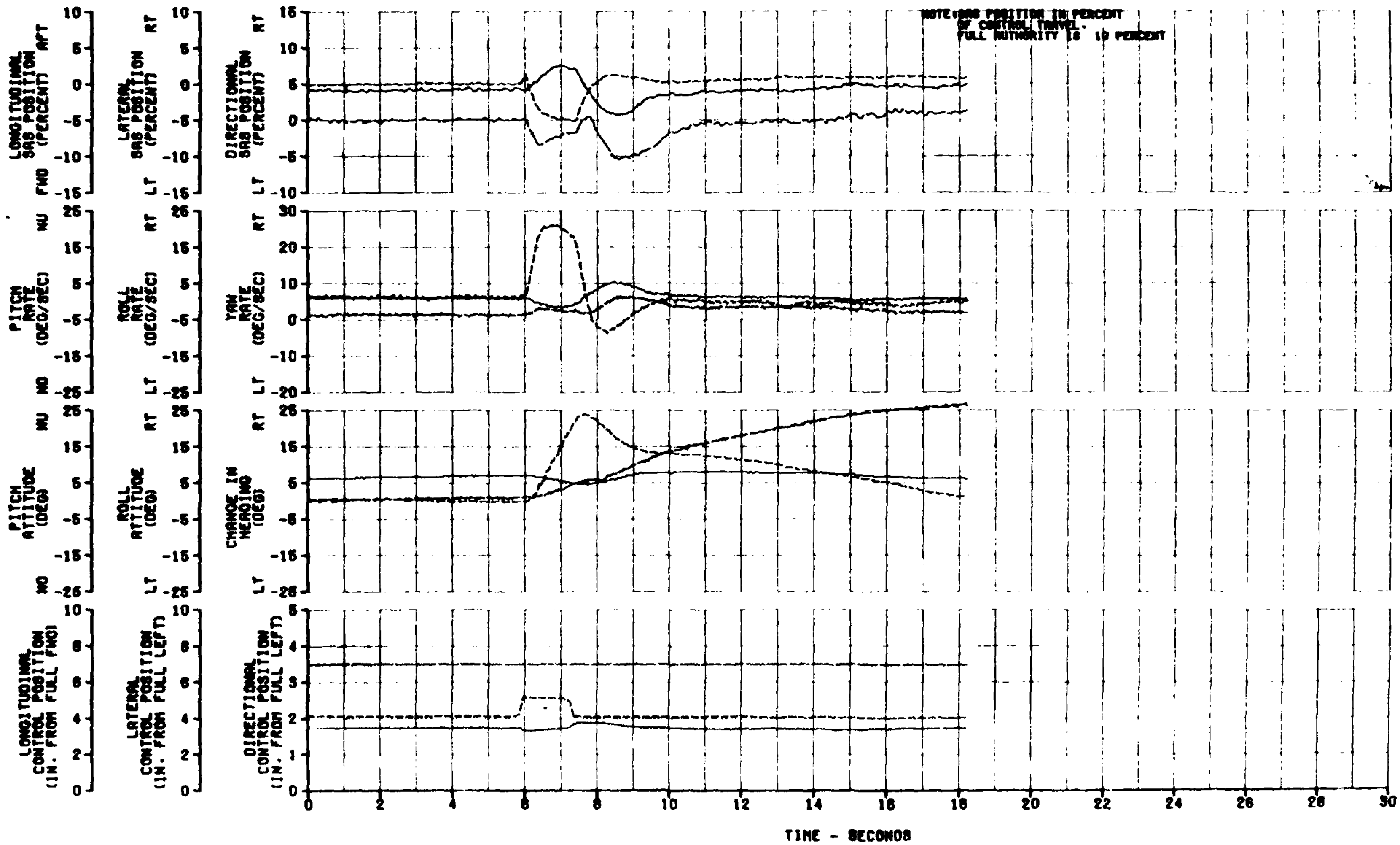


FIGURE 88

RIGHT LATERAL SHORT TERM RESPONSE

YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14220.	207.1 (AFT)	-.1 (LT)	6300.	25.0	288.	91.	OFF

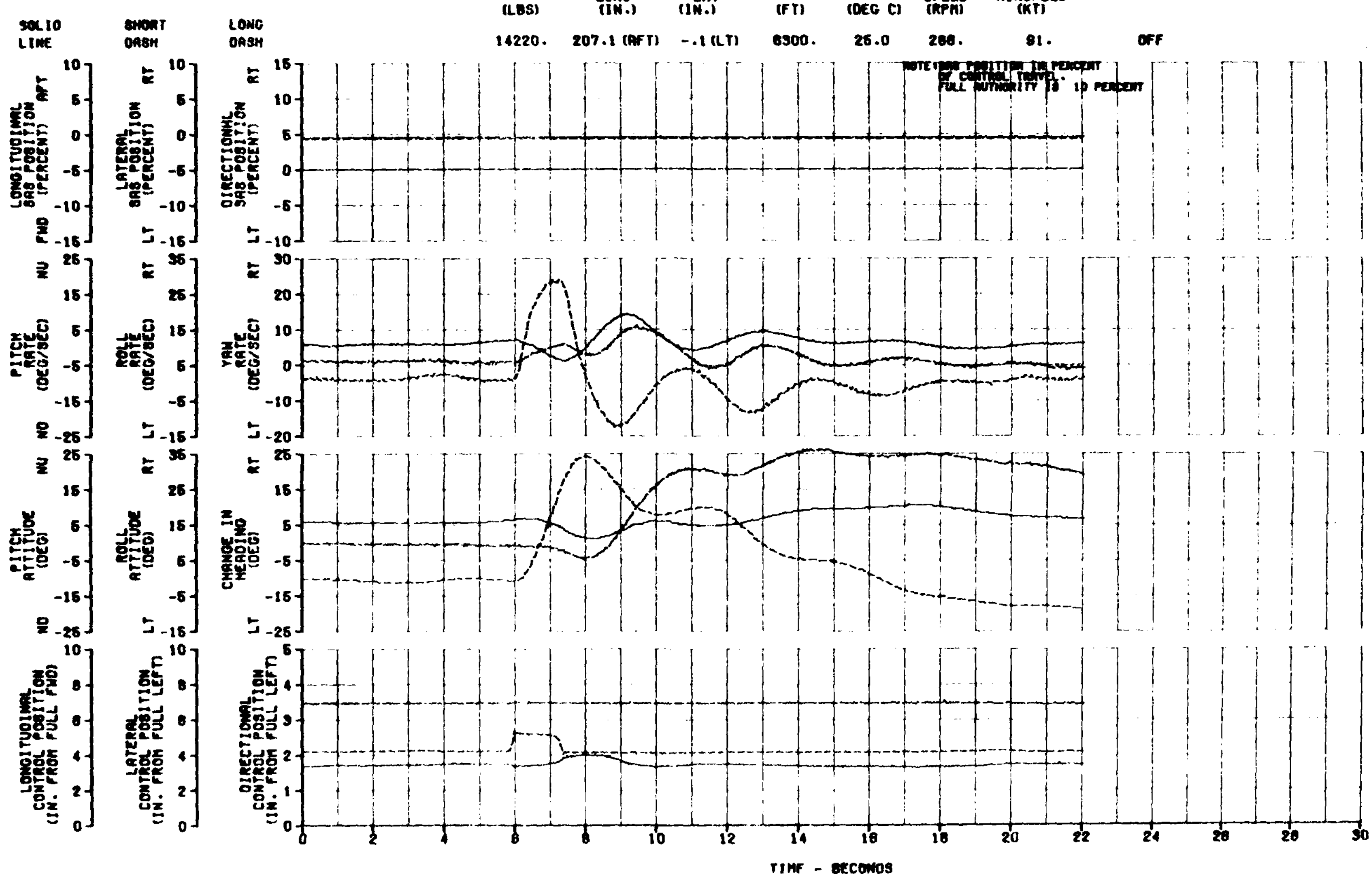


FIGURE 69
LEFT DIRECTIONAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14370.	206.9 (AFT)	-.1 (LT)	5680.	14.5	289.	107.	ON

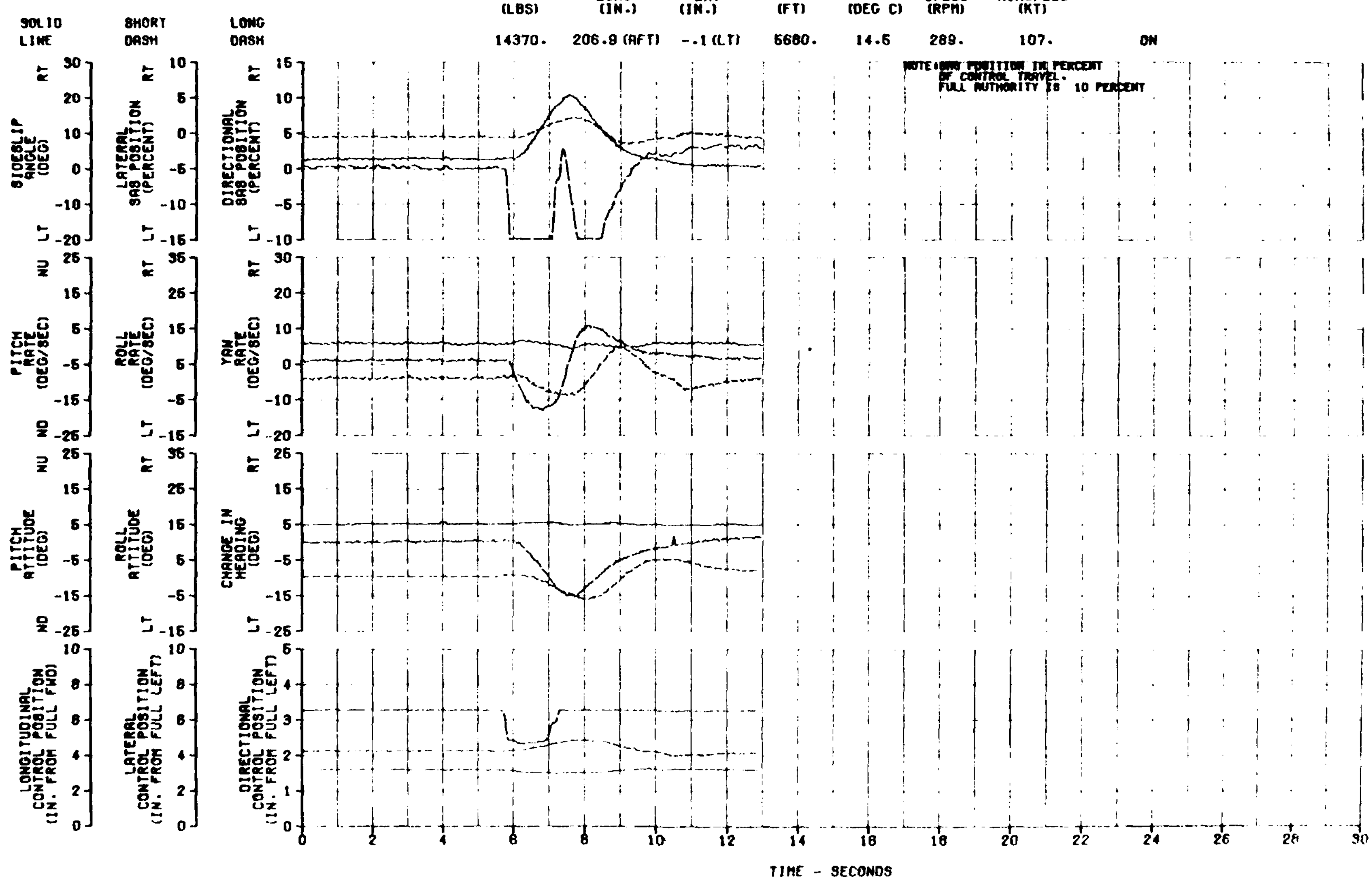


FIGURE 70
RIGHT DIRECTIONAL SHORT TERM RESPONSE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LBS)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
14380.	206.8 (AFT)	-.1 (LT)	6560.	24.0	289.	59.	ON

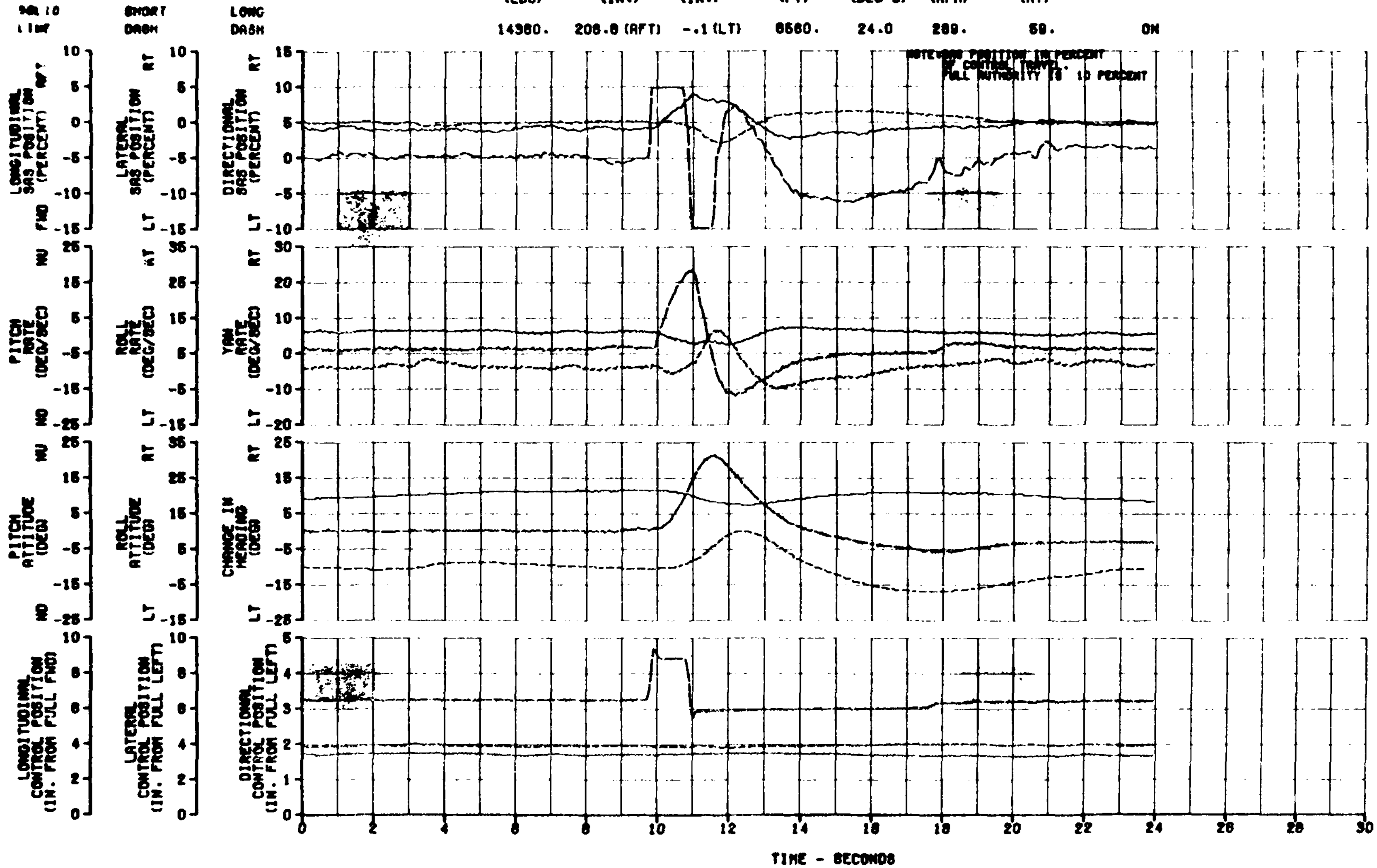


FIGURE 71
RELEASE FROM STEADY SIDESLIP
YAH-64 USA S/N 74-22248

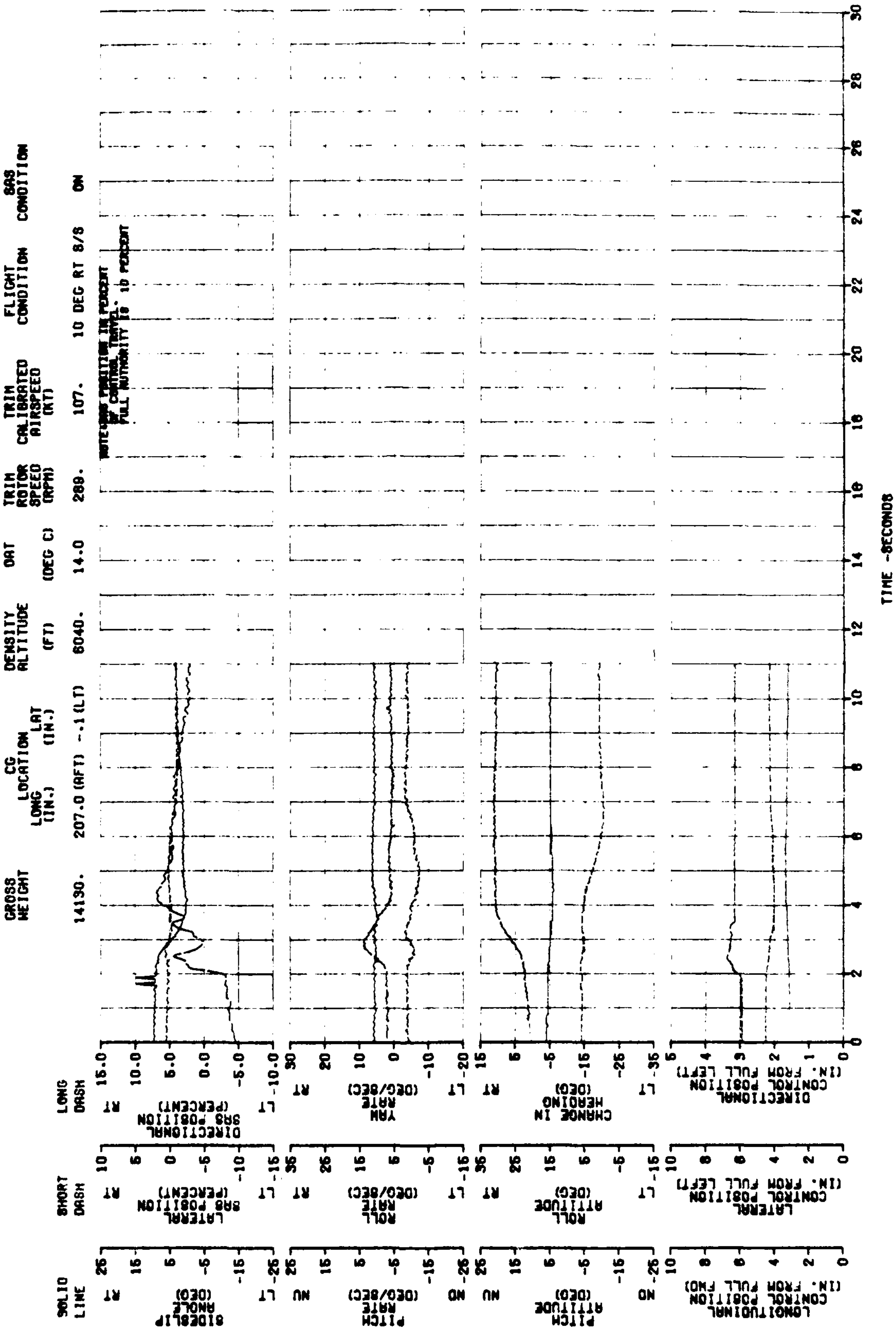
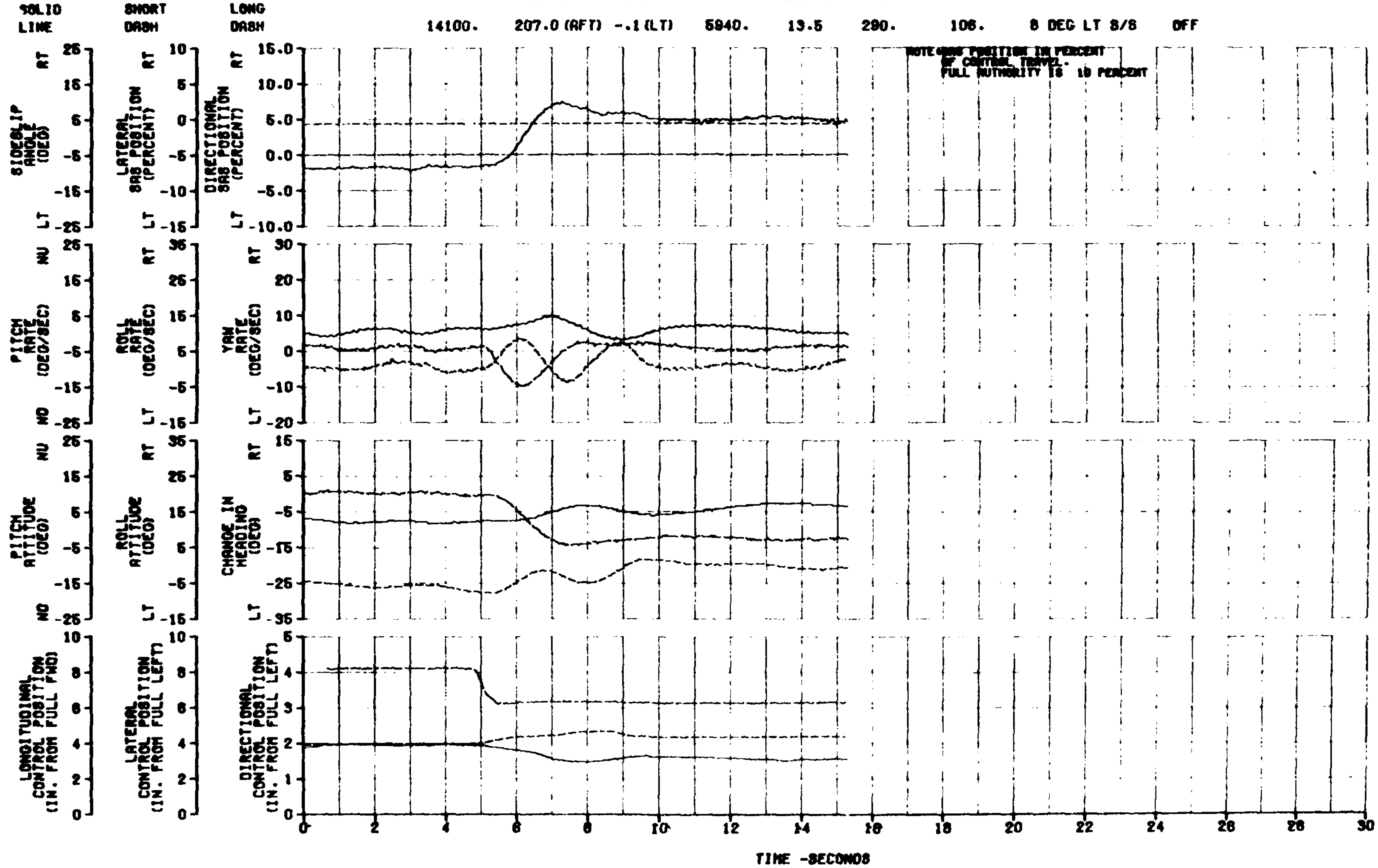
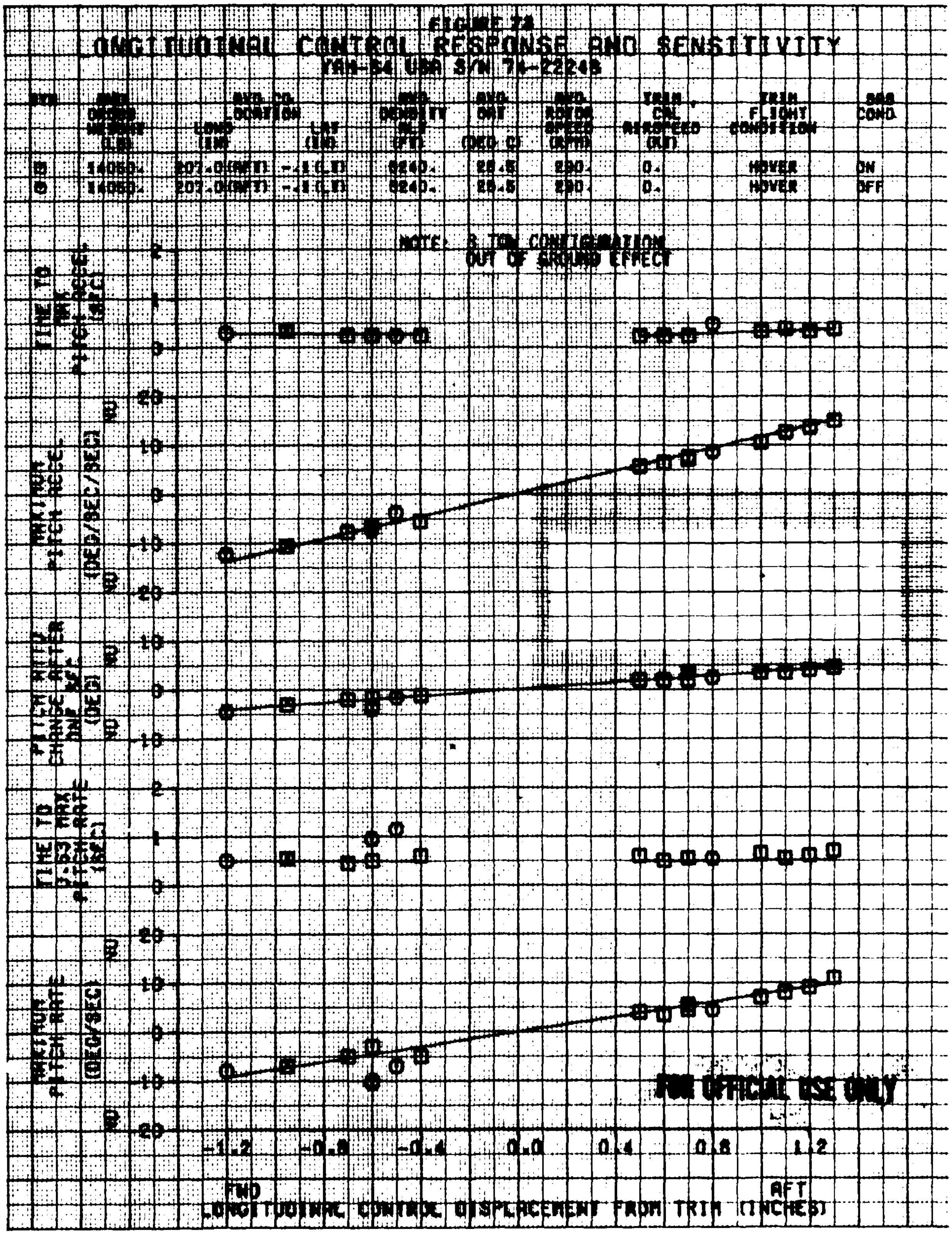


FIGURE 72
RELEASE FROM STEADY SIDESLIP
YAH-64 USA S/N 74-22248

GROSS WEIGHT	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	SAS CONDITION
14100.	207.0 (AFT)	-.1 (LT)	5940.	13.5	290.	106.	8 DEG LT S/S	OFF




LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248
SAS COND
TRIM FLIGHT CONDITION
14060. 207.0(AFT) 8240. 26.5 290. 0. HOVER ON
14060. 207.0(AFT) 8240. 26.5 290. 0. HOVER OFF
NOTE: 8 TOW CONFIGURATION
OUT OF GROUND EFFECT
TIME TO MAX PITCH ACCEL (SEC)
MAXIMUM PITCH ACCEL (DEG/SEC/SEC)
PITCH ATTITUDE CHANGE AFTER ONE SEC (DEG)
TIME TO 0.63 MAX PITCH RATE (SEC)
MAXIMUM PITCH RATE (DEG/SEC)
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-1.2 -0.8 -0.4 0.0 0.4 0.8 1.2
FWD AFT
LONGITUDINAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

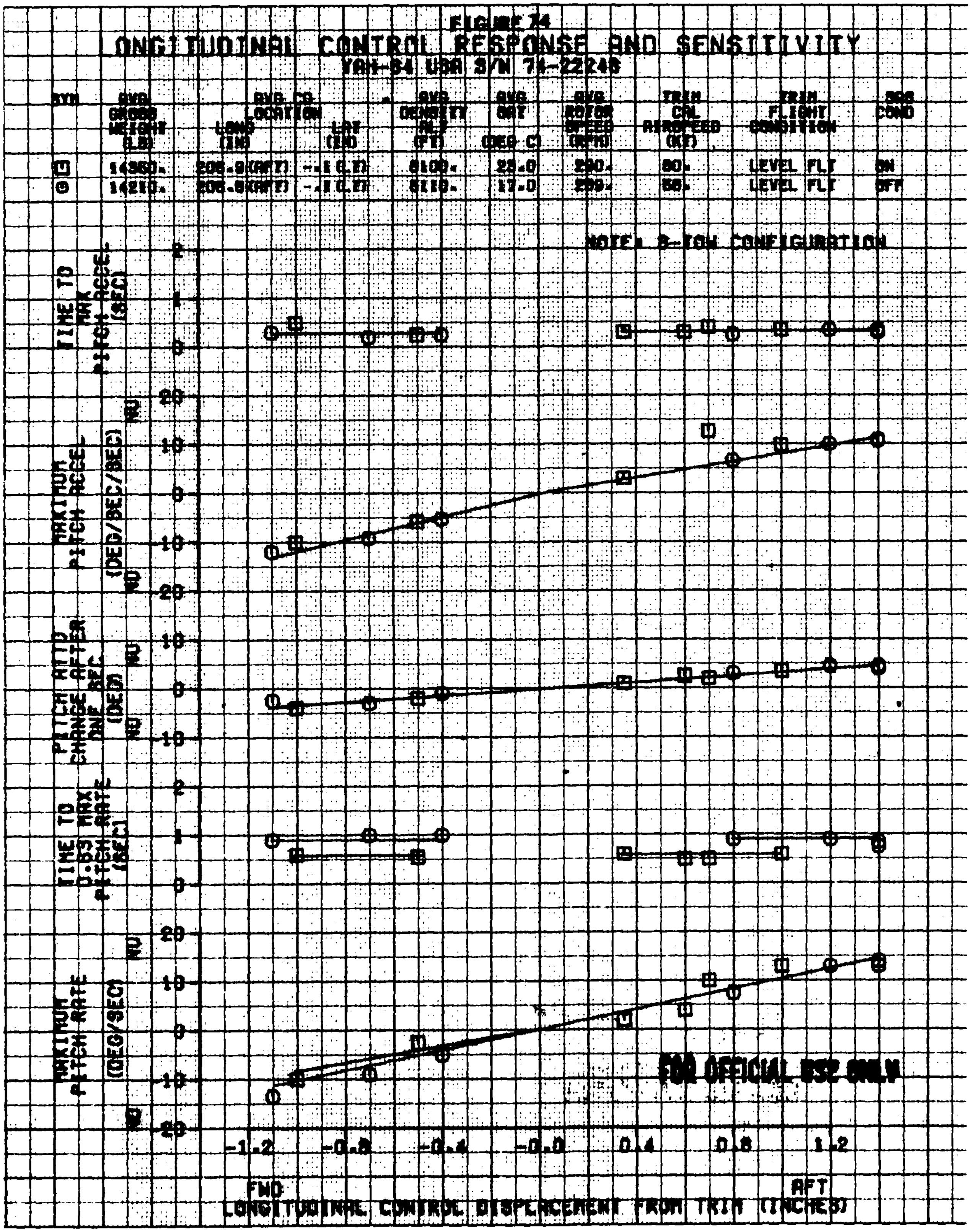

FIGURE 74
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248
SYM | AVG GROSS WEIGHT (LB) | AVG CG LOCATION LONG (IN) | LAT (IN) | AVG DENSITY ALT (FT) | AVG OAT (DEG C) | AVG ROTOR SPEED (RPM) | TRIM CAL AIRSPEED (KT) | TRIM FLIGHT CONDITION | SCAS COND
□ 14360. 206.9(AFT) -.1 (LT) 6100. 23.0 290. 80. LEVEL FLT ON
○ 14210. 206.8(AFT) -.1 (LT) 6110. 17.0 289. 56. LEVEL FLT OFF
NOTE: 8-TOW CONFIGURATION
TIME TO MAX PITCH ACCEL (SEC)
MAXIMUM PITCH ACCEL (DEG/SEC/SEC)
NU
ND
PITCH ATTD CHANGE AFTER ONE SEC (DEG)
NU
ND
TIME TO 0.63 MAX PITCH RATE (SEC)
MAXIMUM PITCH RATE (DEG/SEC)
NU
ND
2
1
0
20
10
0
-10
-20
10
0
-10
2
1
0
20
10
0
-10
-20
-1.2
-0.8
-0.4
-0.0
0.4
0.8
1.2
FWD
AFT
LONGITUDINAL CONTROL DISPLACEMENT FROM TRIM (INCHES)
FOR OFFICIAL USE ONLY

FIGURE 35
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
□	14440.	207.0(AFT)	-.1(LT)	5800.	25.0	290.	90.	LEVEL FLT	ON
○	14440.	207.0(AFT)	-.1(LT)	5800.	25.0	290.	90.	LEVEL FLT	OFF

NOTE: 8 TOW CONFIGURATION

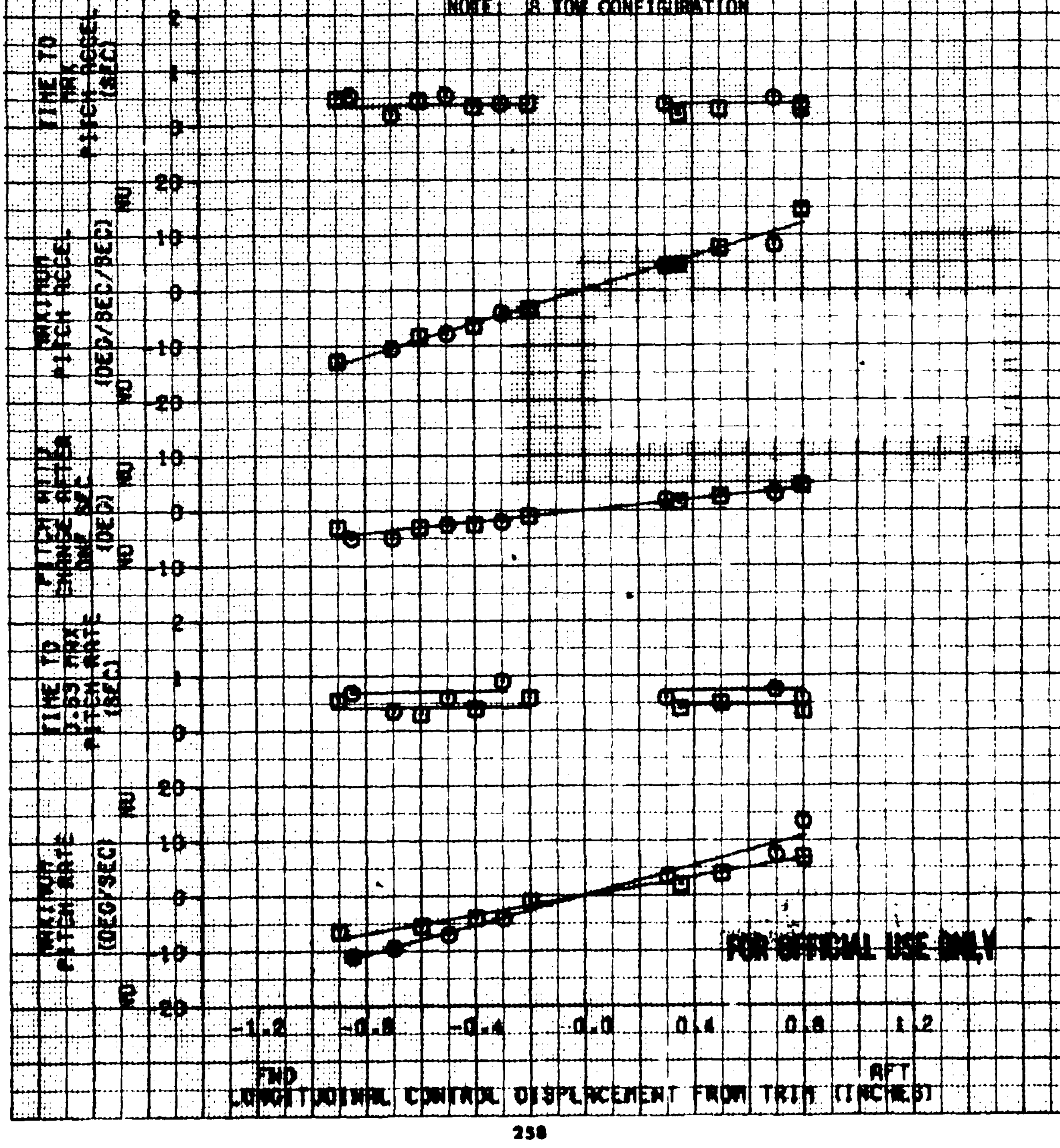

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FIGURE 76

LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
□	14570.	206.8(AFT)	-.1(LT)	6000.	22.5	288.	110.	LEVEL FLT	ON
○	14570.	206.8(AFT)	-.1(LT)	6000.	22.5	288.	110.	LEVEL FLT	OFF

NOTE: B TOW CONFIGURATION

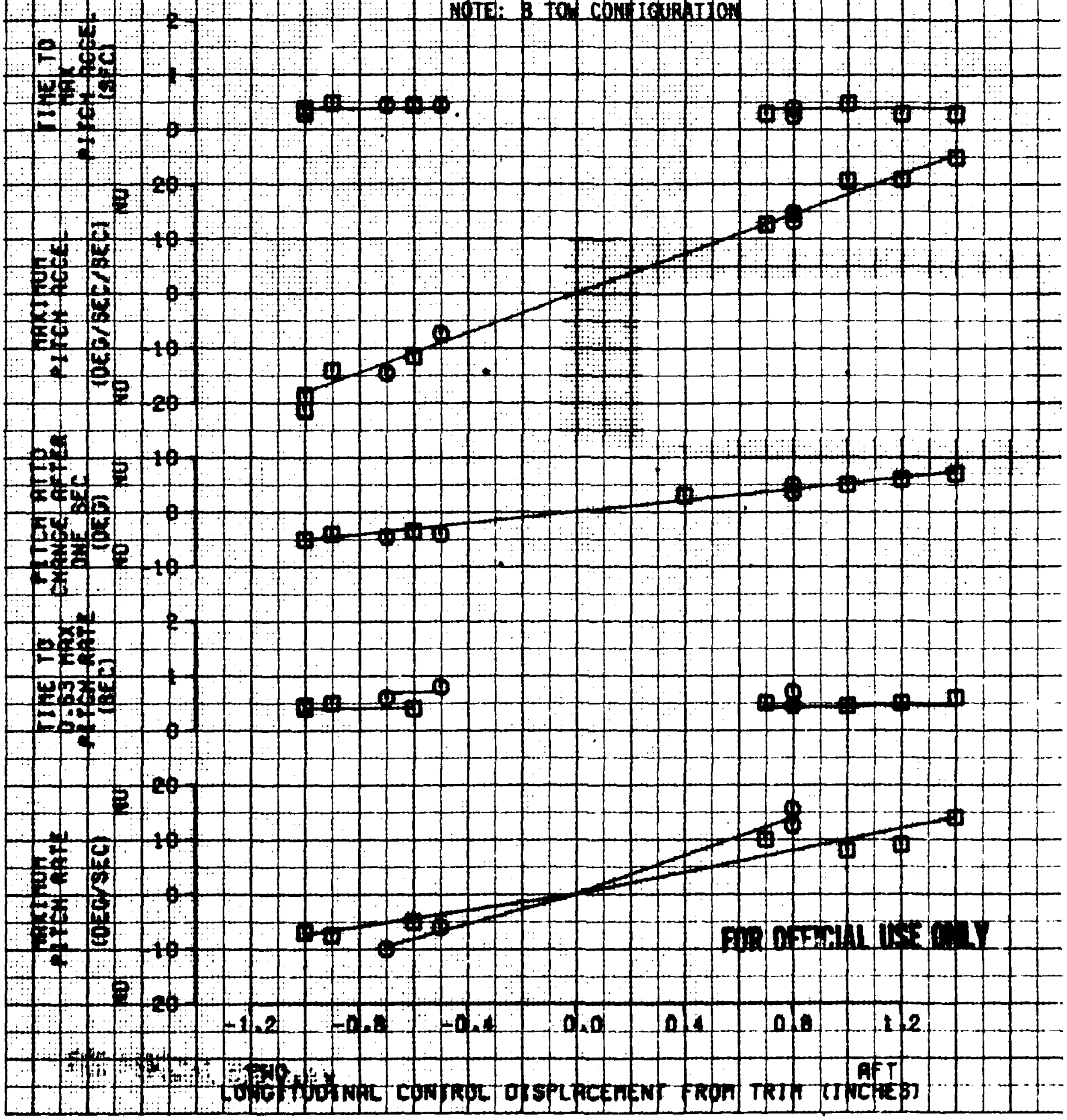

FIGURE 77
LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
○	14050.	207.0(AFT)	-.1(LT)	6240.	26.5	290.	0.	HOVER	ON
□	14050.	207.0(AFT)	-.1(LT)	6240.	26.5	290.	0.	HOVER	OFF

NOTE: 8 TOW CONFIGURATION
OUT OF GROUND EFFECT

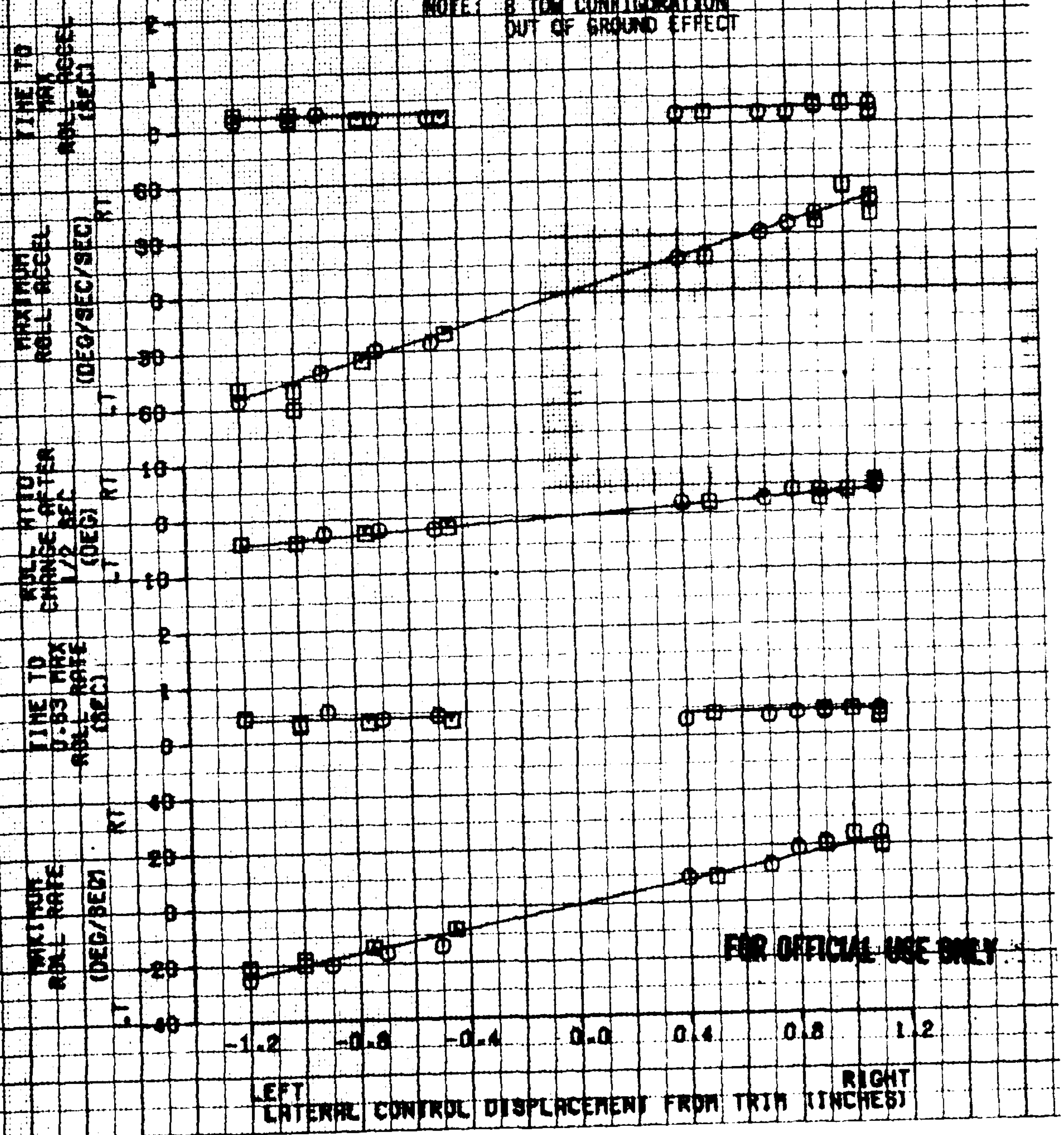

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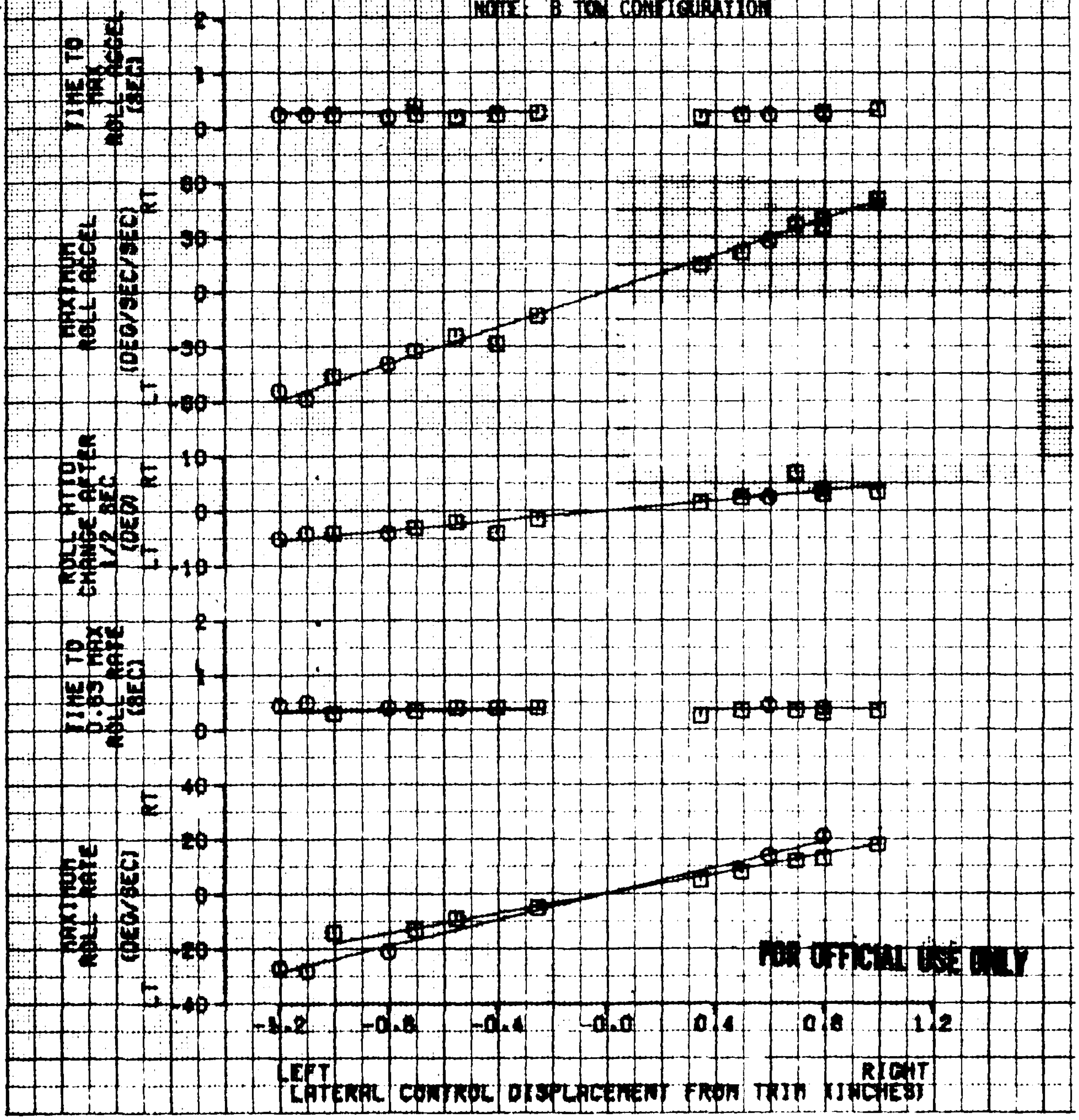
FIGURE 78

LATERAL CONTROL RESPONSE AND SENSITIVITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
○	14360.	200.9(AFT)	-.1(LT)	6100.	23.0	290.	80.	LEVEL FLT	ON
□	14210.	200.8(AFT)	-.1(LT)	5510.	17.0	289.	80.	LEVEL FLT	OFF

NOTE: 8 TOW CONFIGURATION



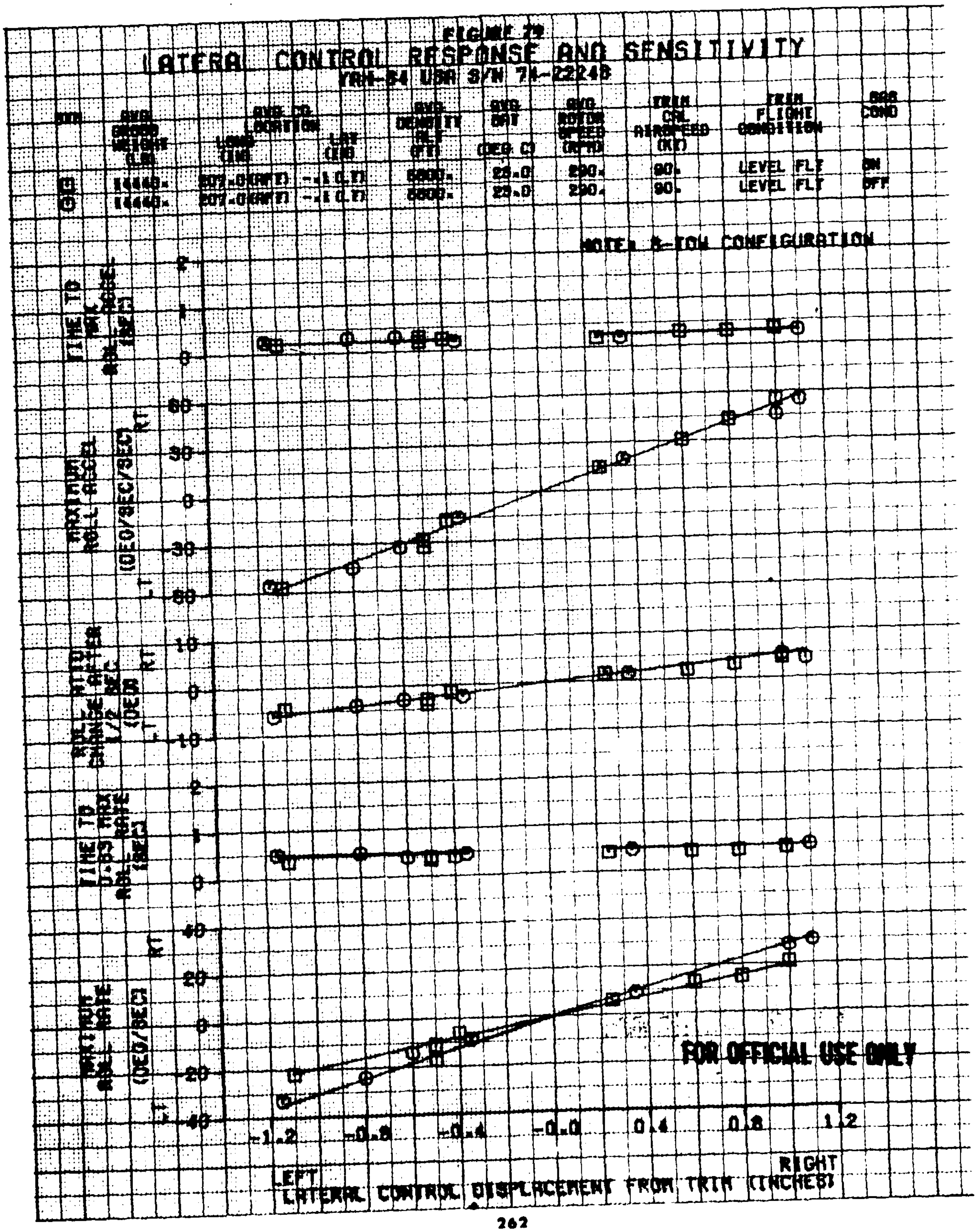


FIGURE 79
LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN)
LAT (IN)
AVG DENSITY ALT (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
DAS COND
14440. 207.0(AFT) -.1(LT) 5800. 25.0 290. 90. LEVEL FLT ON
14440. 207.0(AFT) -.1(LT) 5800. 25.0 290. 90. LEVEL FLT OFF
NOTE: 8-TOW CONFIGURATION
TIME TO MAX ROLL ACCEL (SEC)
MAXIMUM ROLL ACCEL (DEG/SEC/SEC)
ROLL ATTITUDE CHANGE AFTER 1/2 SEC (DEG)
TIME TO 0.63 MAX ROLL RATE (SEC)
MAXIMUM ROLL RATE (DEG/SEC)
RT
LT
LEFT
RIGHT
LATERAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

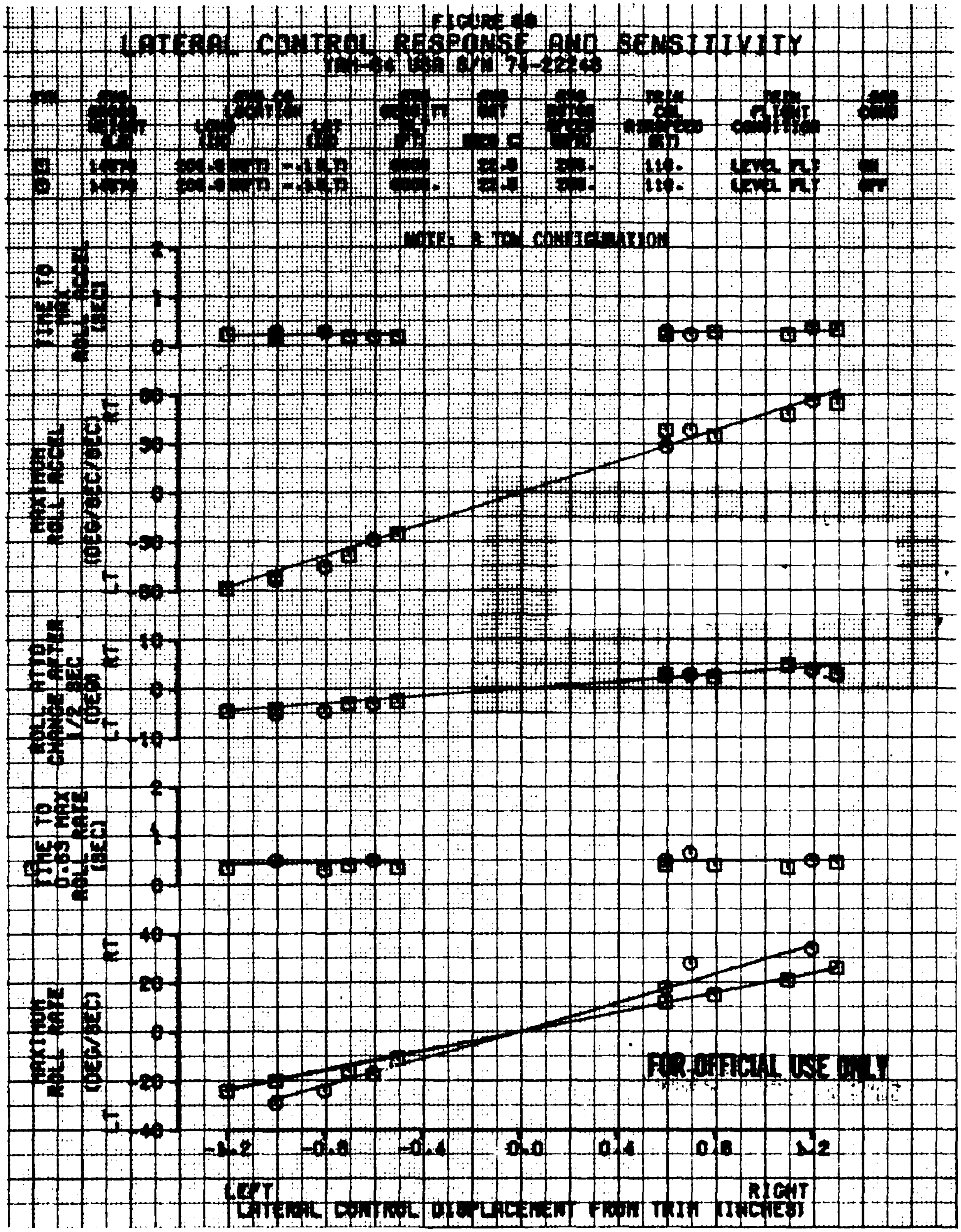
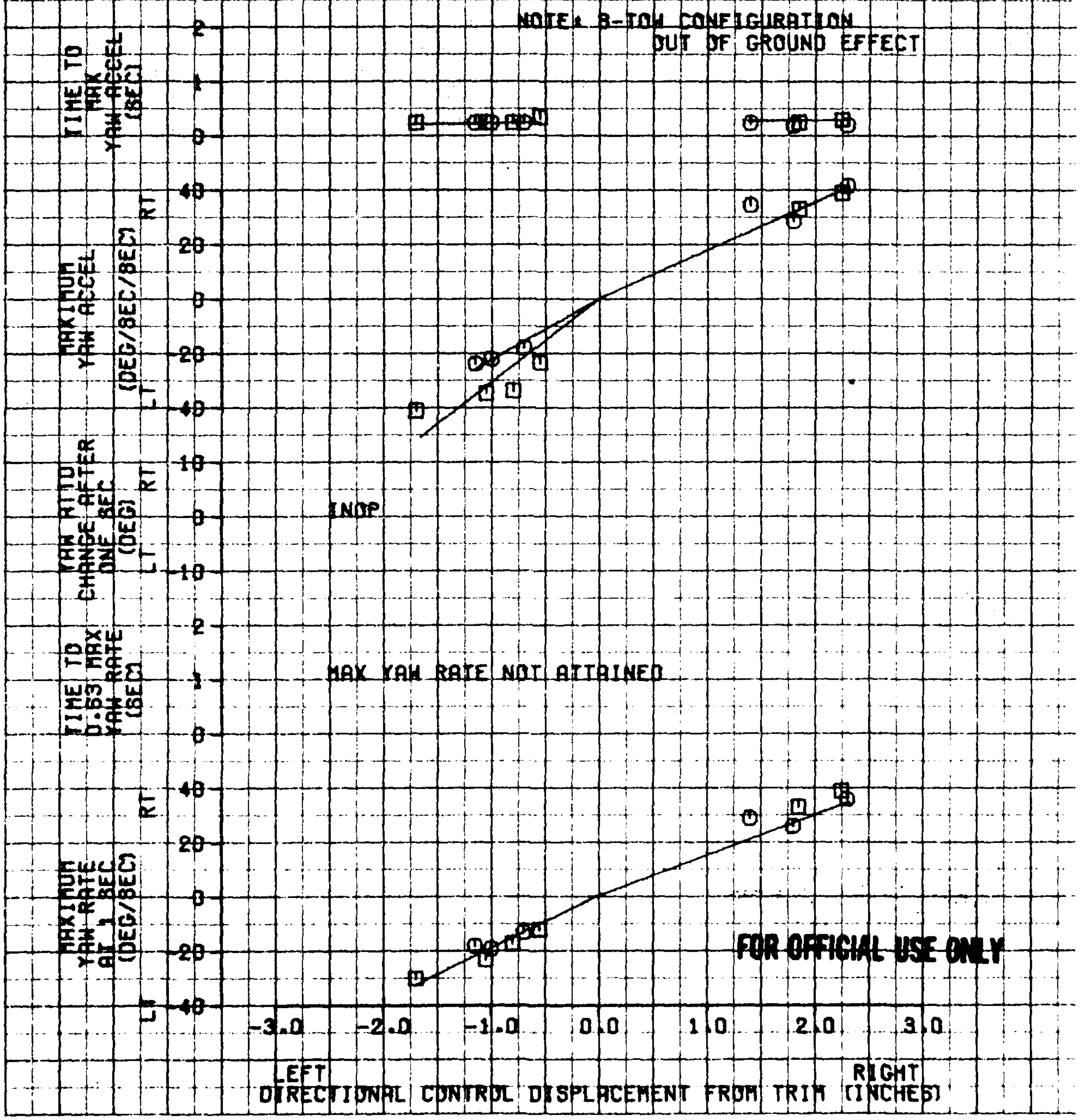

LATERAL CONTROL RESPONSE AND SENSITIVITY
LEVEL FLT
LEVEL FLT
NOTE: 3 TOW CONFIGURATION
TIME TO MAX ROLL ACCEL (SEC)
MAXIMUM ROLL ACCEL (DEG/SEC/SEC)
ROLL ATTID CHANGE AFTER 1/2 SEC (DEG)
TIME TO 0.63 MAX ROLL RATE (SEC)
MAXIMUM ROLL RATE (DEG/SEC)
RT
LT
FOR OFFICIAL USE ONLY
-1.2
-0.8
-0.4
0.0
0.4
0.8
1.2
LEFT
RIGHT
LATERAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

FIGURE 81
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
□	14050.	207.0(AFT)	-.1(LT)	6240.	28.5	290.	0.	HOVER	ON
○	14050.	207.0(AFT)	-.1(LT)	6240.	28.5	290.	0.	HOVER	OFF



FOR OFFICIAL USE ONLY

FIGURE 82

DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY

YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
☐	14360.	200.0(FWD)	-.1(LT)	11000.	10.6	289.	0.	HOVER	ON

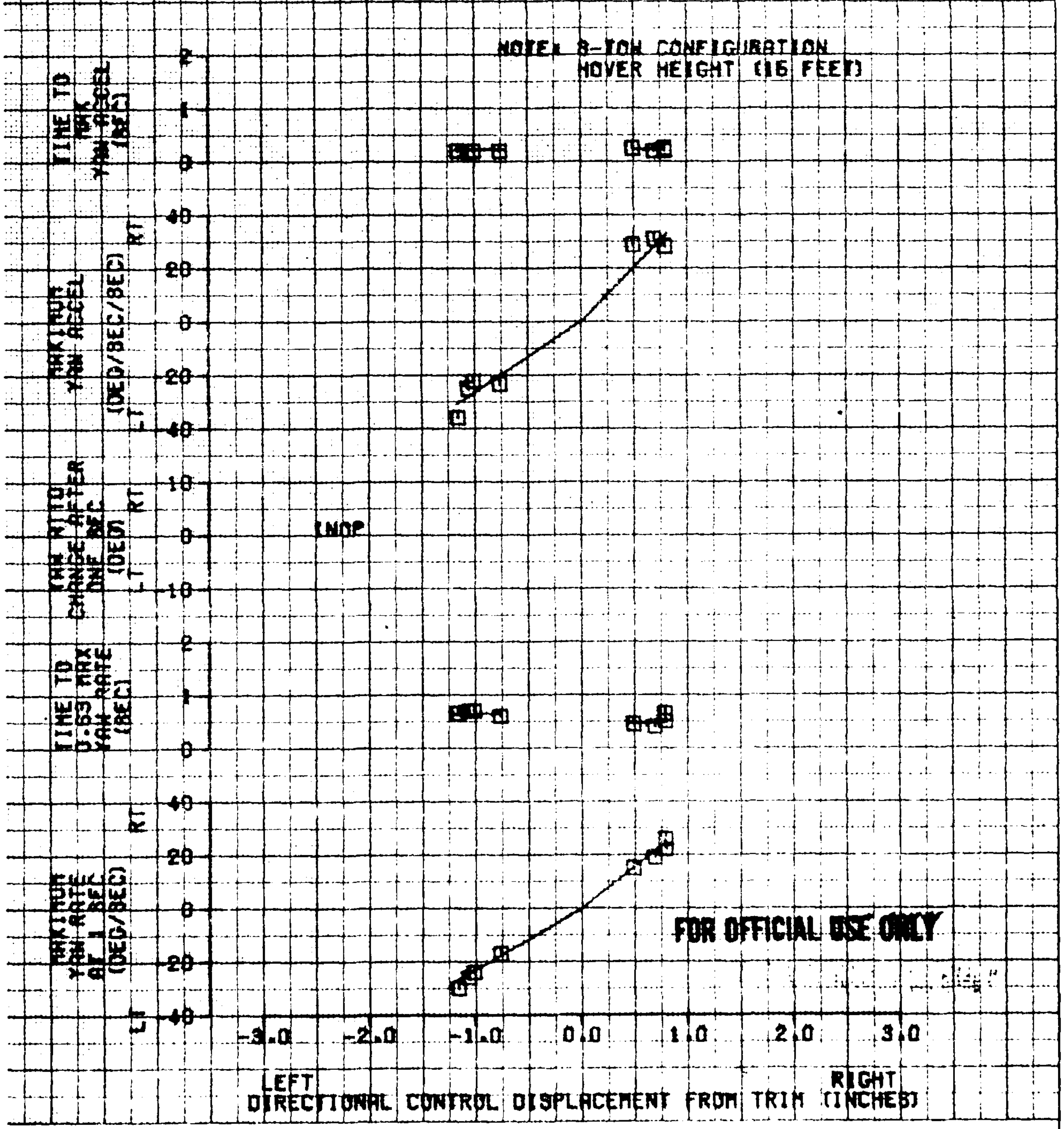
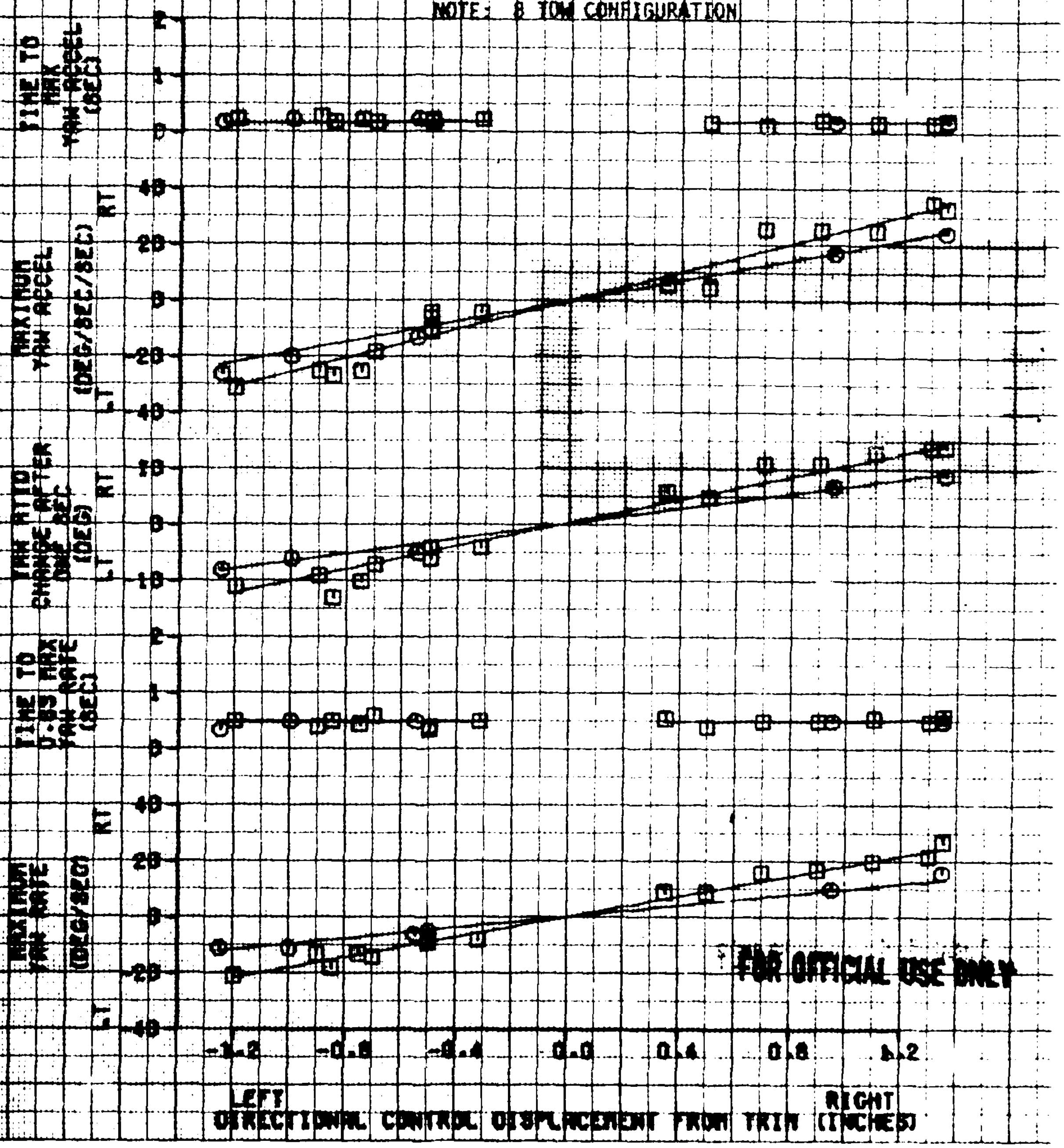


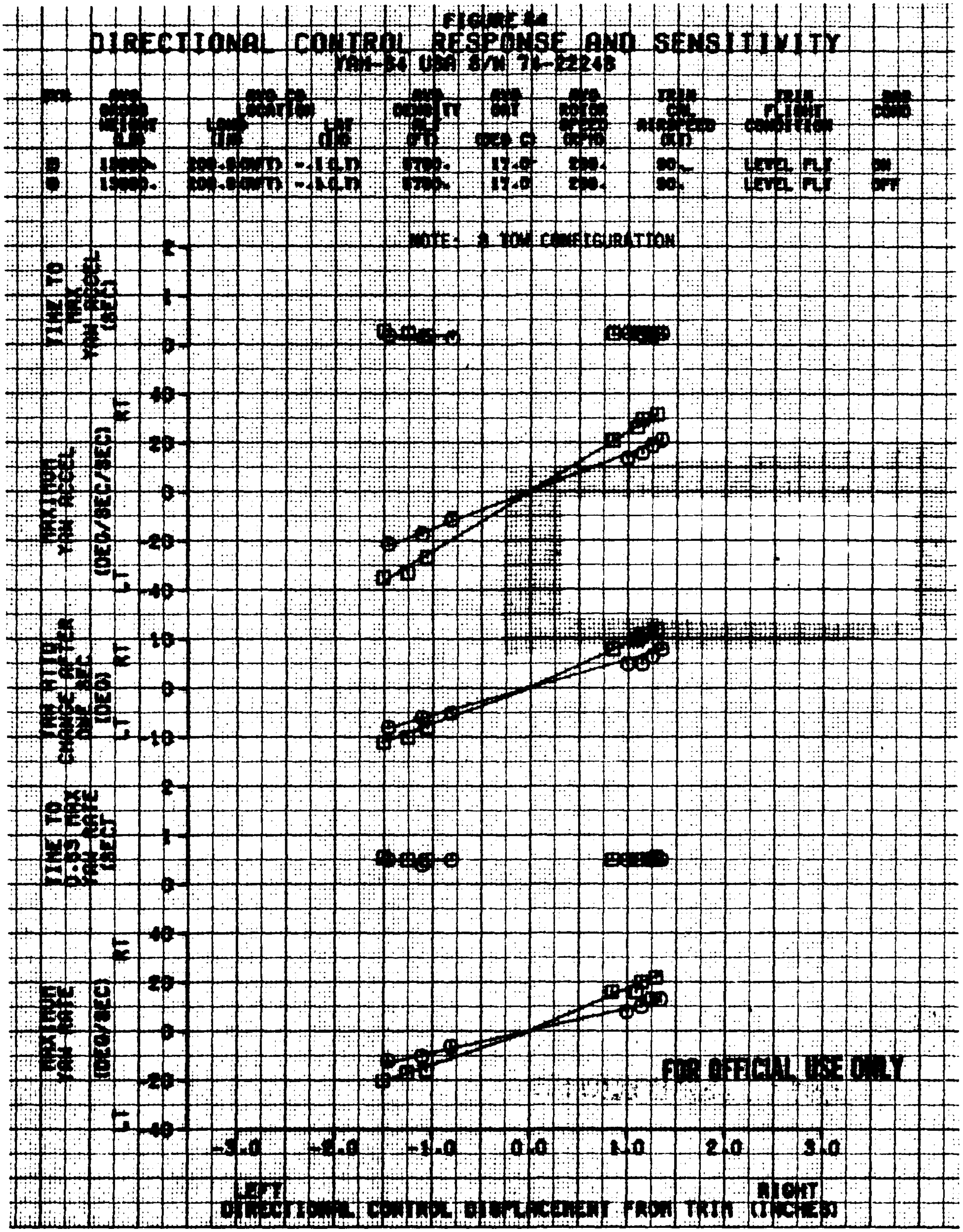
FIGURE 83
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS COND
□	14300.	200.2 (AFT)	-.1 (LT)	8100.	23.0	290.	80.	LEVEL FLT	ON
○	14440.	200.2 (AFT)	-.3 (LT)	5470.	24.0	290.	80.	LEVEL FLT	OFF

NOTE: 8 TOW CONFIGURATION



FOR OFFICIAL USE ONLY


DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
LEVEL FLT
LEVEL FLT
FOR OFFICIAL USE ONLY
-1.0
0.0
1.0
2.0
3.0
LEFT
RIGHT

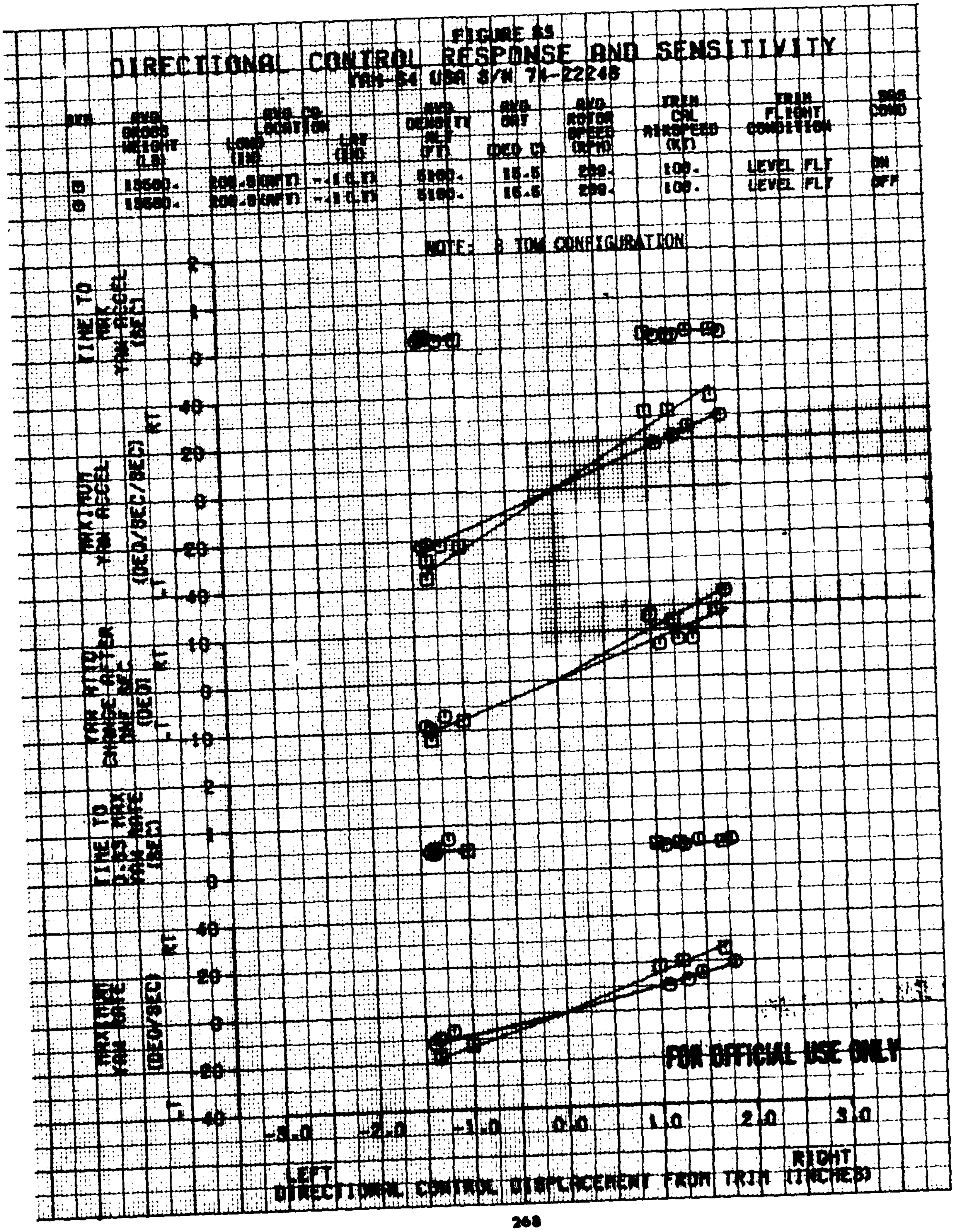


FIGURE
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
(DEG C)
15500.
5100.
15.5
100.
LEVEL FLT
ON
OFF
NOTE: 8 TOW CONFIGURATION
TIME TO MAX YAW ACCEL (SEC)
MAXIMUM YAW ACCEL (DEG/SEC/SEC)
TIME TO 0.63 MAX YAW RATE (SEC)
MAXIMUM YAW RATE (DEG/SEC)
FOR OFFICIAL USE ONLY
-3.0
-2.0
-1.0
0.0
1.0
2.0
3.0
LEFT
RIGHT
DIRECTIONAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

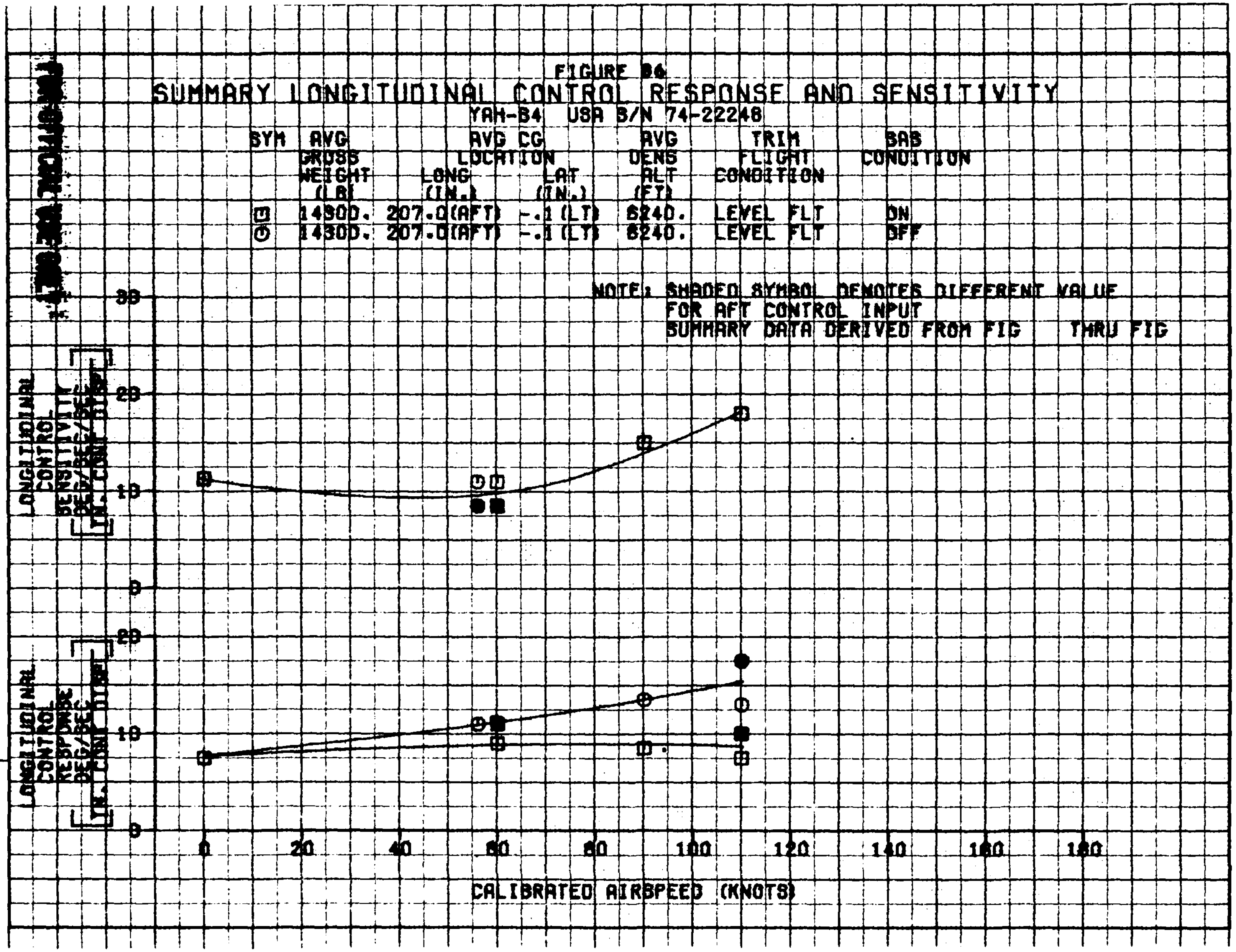


FIGURE 86
SUMMARY LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248
SYM | AVG GROSS WEIGHT (LB) | AVG CG LOCATION LONG (IN.) | LAT (IN.) | AVG DENS ALT (FT) | TRIM FLIGHT CONDITION | SAS CONDITION
14300. | 207.0(AFT) | -.1(LT) | 6240. | LEVEL FLT | ON
14300. | 207.0(AFT) | -.1(LT) | 6240. | LEVEL FLT | OFF
NOTE: SHADED SYMBOL DENOTES DIFFERENT VALUE
FOR AFT CONTROL INPUT
SUMMARY DATA DERIVED FROM FIG THRU FIG
LONGITUDINAL CONTROL SENSITIVITY [DEG/SEC/SEC / IN. CONT DISPL]
LONGITUDINAL CONTROL RESPONSE [DEG/SEC / IN. CONT DISPL]
CALIBRATED AIRSPEED (KNOTS)
0
10
20
30
0
20
40
60
80
100
120
140
160
180

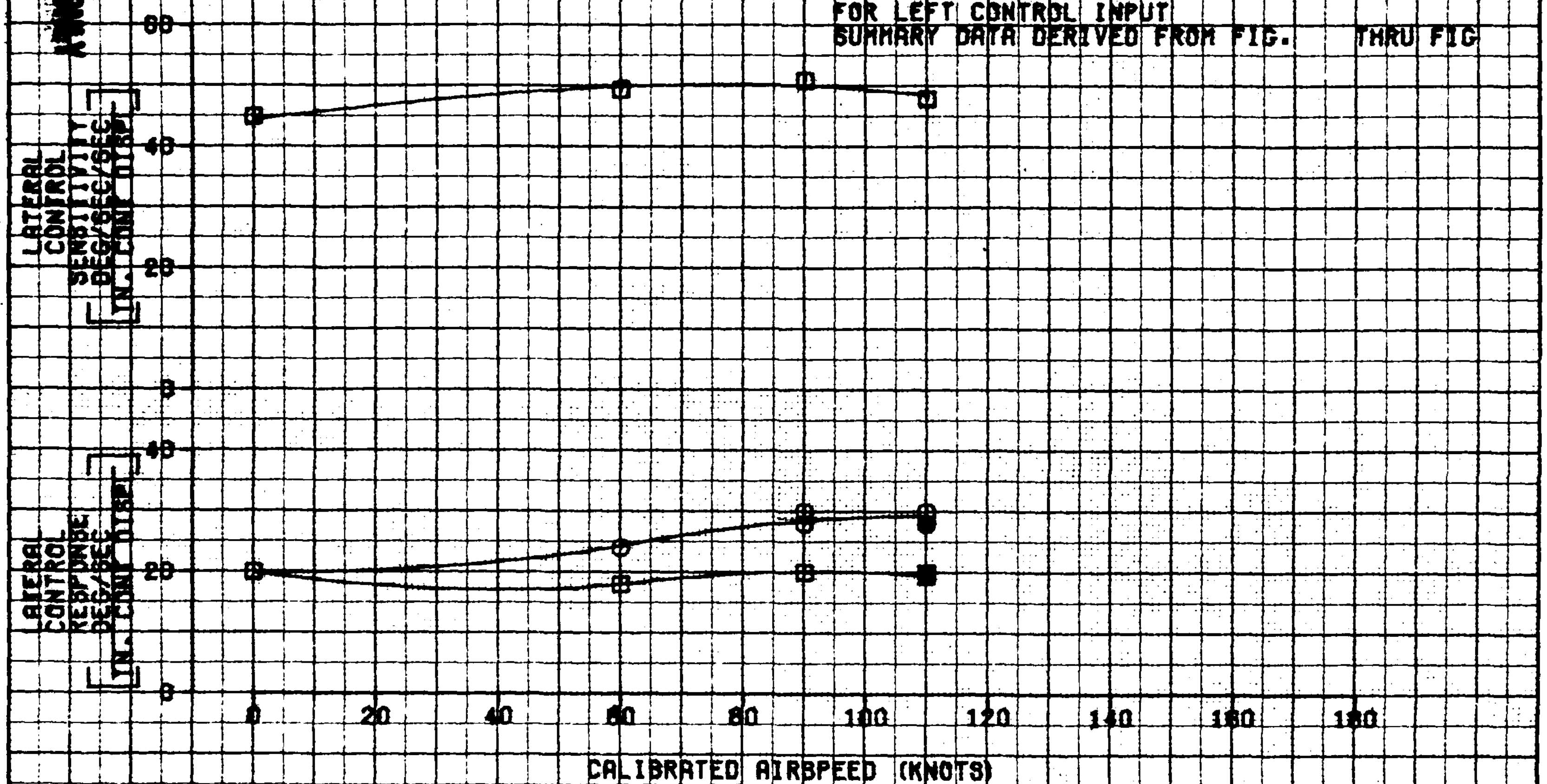
FIGURE 87

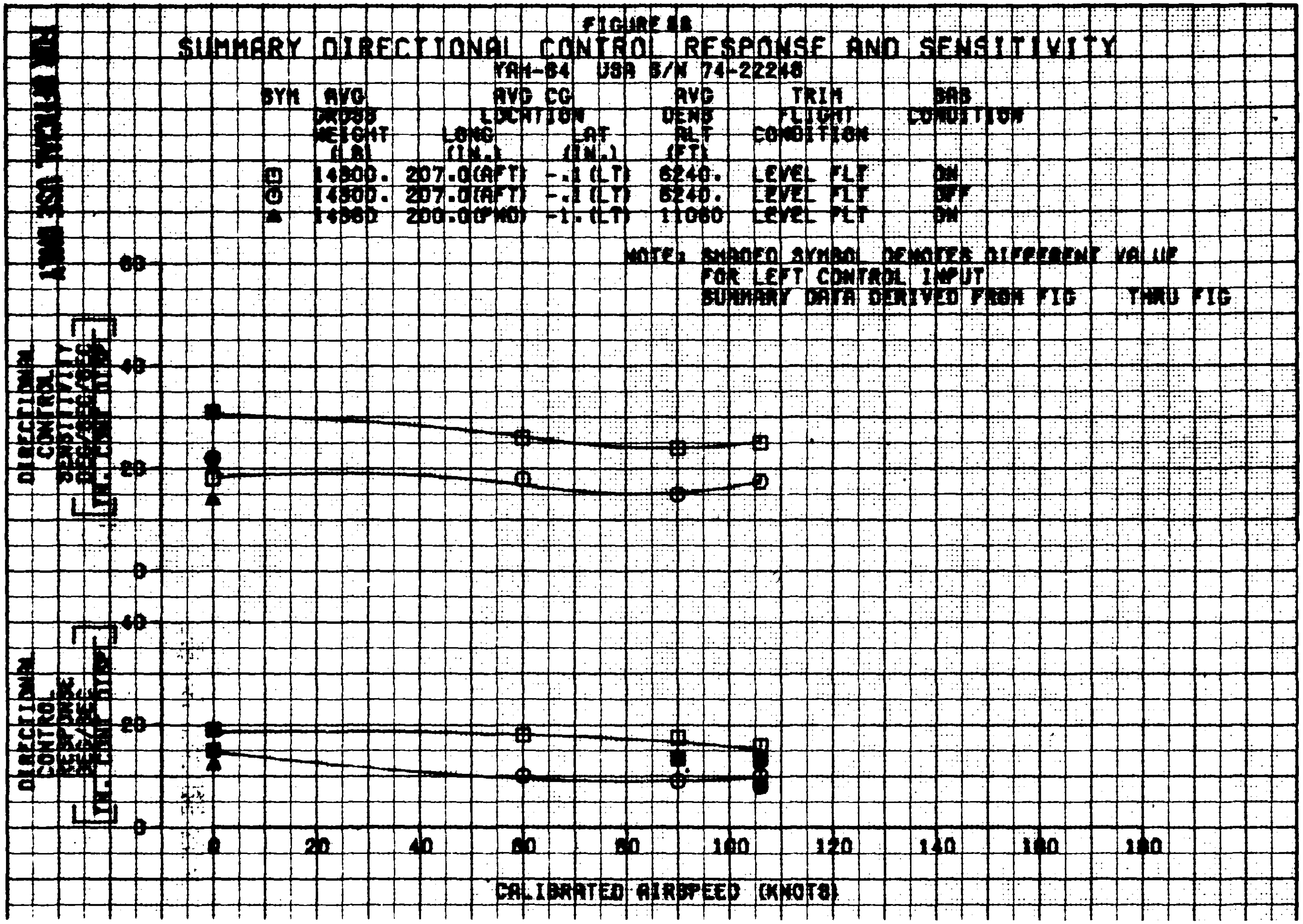
SUMMARY LATERAL CONTROL RESPONSE AND SENSITIVITY

YAH-64 USA S/N 74-22248

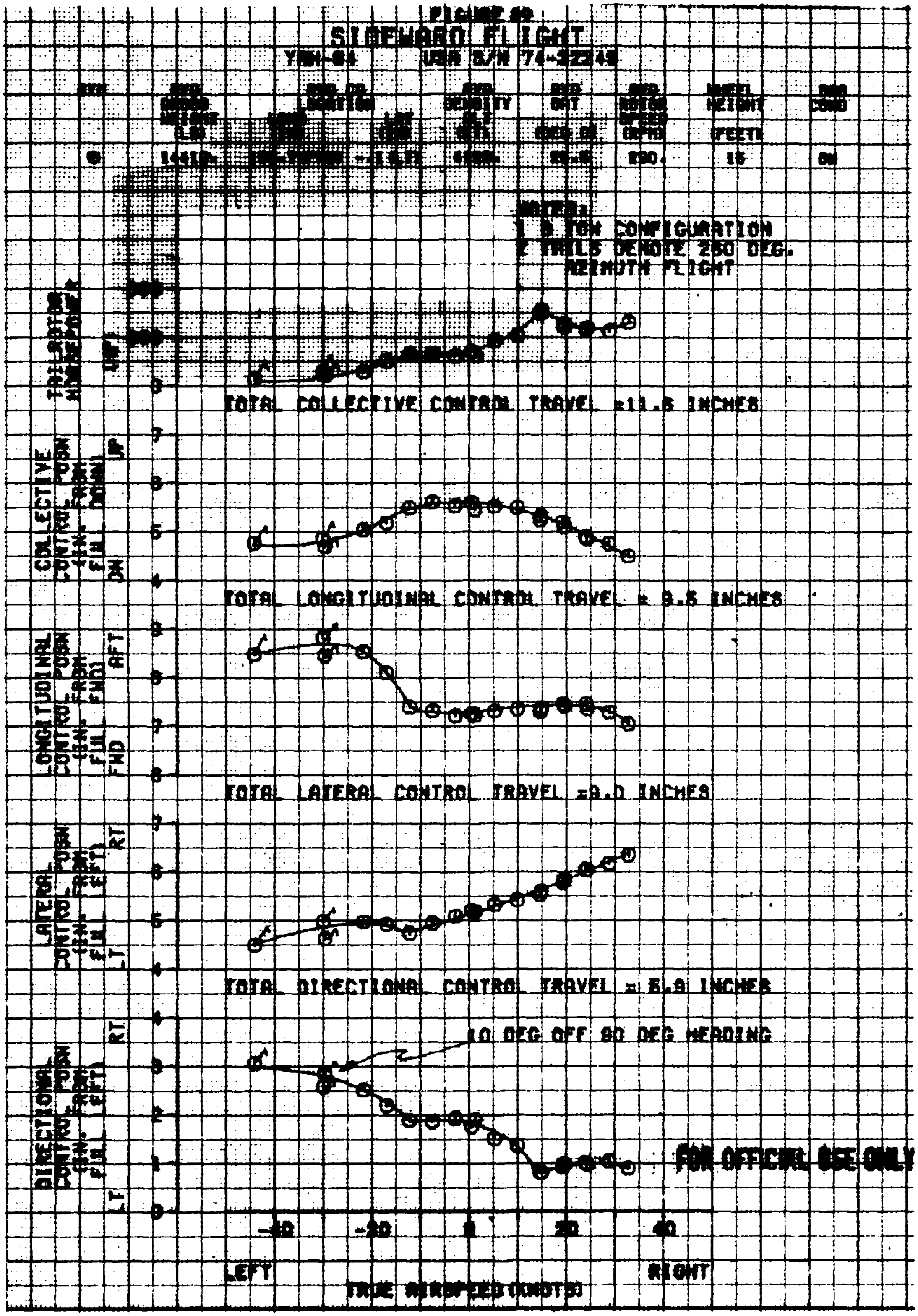
SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	TRIM FLIGHT CONDITION	SAS CONDITION
□	14300.	207.0(AFT)	-.1 (LT)	6240.	LEVEL FLT	ON
○	14300.	207.0(AFT)	-.1 (LT)	6240.	LEVEL FLT	OFF

NOTE: SHADED SYMBOL DENOTES DIFFERENT VALUE
FOR LEFT CONTROL INPUT
SUMMARY DATA DERIVED FROM FIG.   THRU FIG

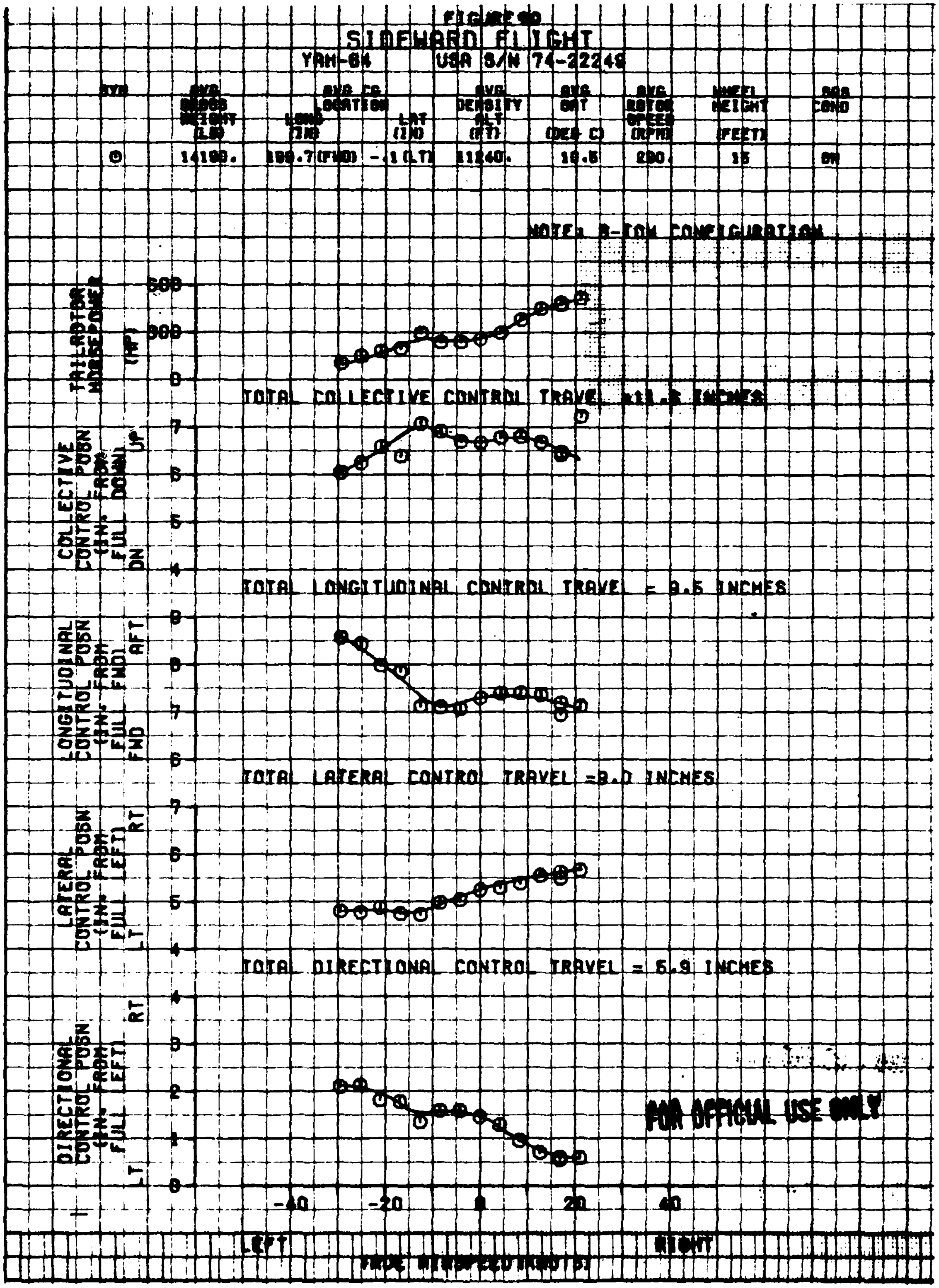




FOR OFFICIAL USE ONLY
FIGURE
SUMMARY DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
YAH-64 USA S/N 74-22248
SYM AVG GROSS WEIGHT (LB) | AVG CG LOCATION LONG (IN.) | LAT (IN.) | AVG DENS ALT (FT) | TRIM FLIGHT CONDITION | SAS CONDITION
14800. 207.0(AFT) -.1(LT) 6240. LEVEL FLT ON
14800. 207.0(AFT) -.1(LT) 6240. LEVEL FLT OFF
14860 200.0(FWD) -1.(LT) 11060 LEVEL FLT ON
NOTE: SHADED SYMBOL DENOTES DIFFERENT VALUE
FOR LEFT CONTROL INPUT
SUMMARY DATA DERIVED FROM FIG THRU FIG
DIRECTIONAL CONTROL SENSITIVITY [DEG/SEC/SEC IN. CONT DISP]
DIRECTIONAL CONTROL RESPONSE [DEG/SEC IN. CONT DISP]
0
20
40
60
80
100
120
140
160
180
CALIBRATED AIRSPEED (KNOTS)



SIDEWARD FLIGHT
USA S/N 74-22249
(FEET)
15
NOTES:
1 D TOW CONFIGURATION
2 TAILS DENOTE 260 DEG. AZIMUTH FLIGHT
TOTAL COLLECTIVE CONTROL TRAVEL =11.5 INCHES
COLLECTIVE CONTROL POSN (IN. FROM FUL DOWN)
UP
DN
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.5 INCHES
LONGITUDINAL CONTROL POSN (IN. FROM FUL FWD)
AFT
FWD
TOTAL LATERAL CONTROL TRAVEL =9.0 INCHES
LATERAL CONTROL POSN (IN. FROM FUL LEFT)
RT
LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.9 INCHES
10 DEG OFF 90 DEG HEADING
DIRECTIONAL CONTROL POSN (IN. FROM FUL LEFT)
RT
LT
-40
-20
0
20
40
LEFT
RIGHT
TRUE AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY

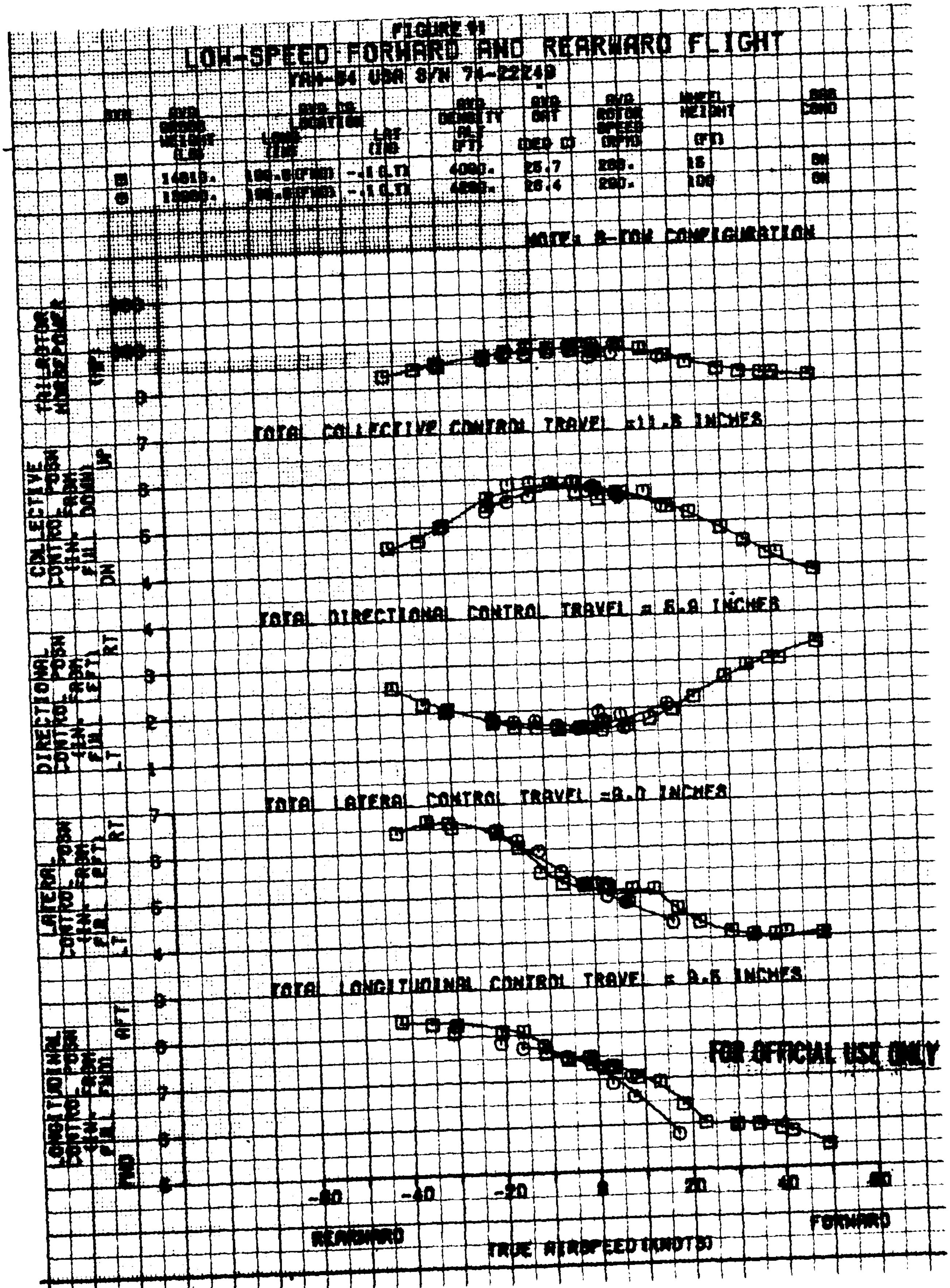


FIGURE 91
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 74-22249
NOTE: 8-TOW CONFIGURATION
TOTAL COLLECTIVE CONTROL TRAVEL =11.8 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.8 INCHES
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.5 INCHES
COLLECTIVE CONTROL POSN
DIRECTIONAL CONTROL POSN
LONGITUDINAL CONTROL POSN
UP
DN
RT
LT
AFT
FWD
FOR OFFICIAL USE ONLY
-60
-40
-20
0
20
40
60
REARWARD
FORWARD
TRUE AIRSPEED (KNOTS)

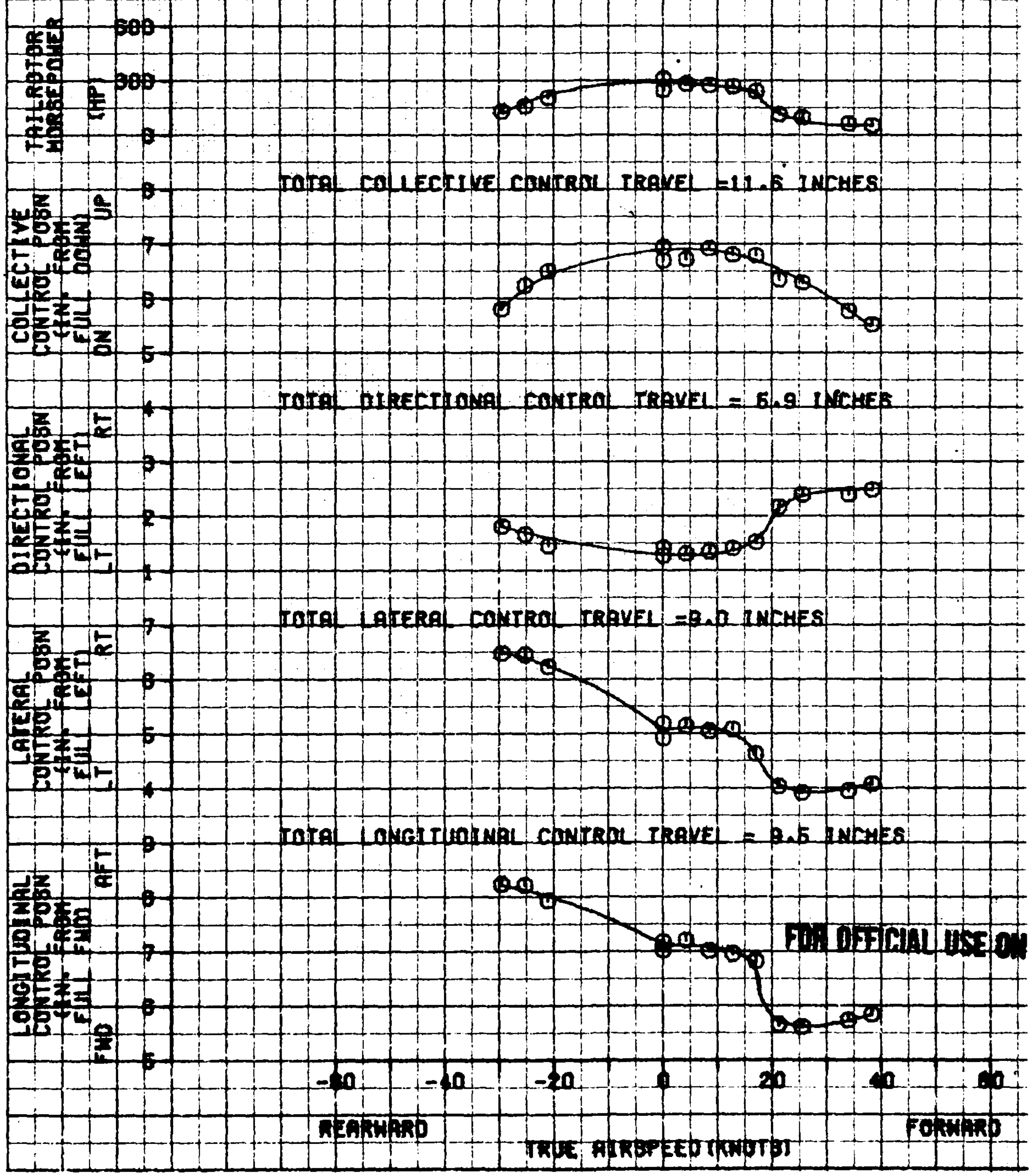
FIGURE 92

LOW-SPEED FORWARD AND REARWARD FLIGHT

YAH-64 USA S/N 74-22249

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)	SAS COND
⊙	14680.	200.0(FWD)	-.1 (LT)	10950.	10.0	290.	15	ON

NOTE: B-TOW CONFIGURATION



FOR OFFICIAL USE ON

FIGURE 93
LATERAL ACCELERATION
YAH-64 USA S/N 74-22249

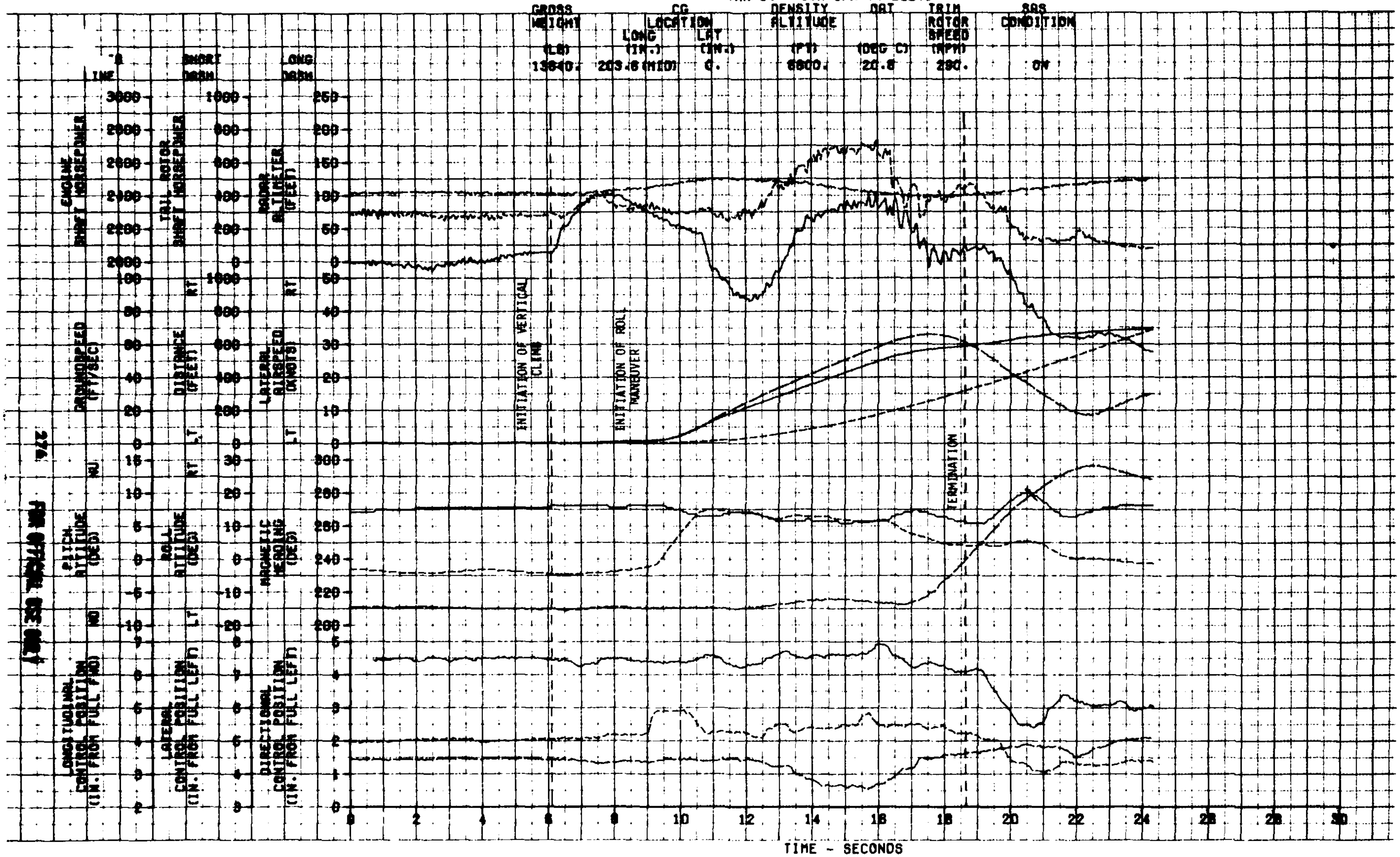
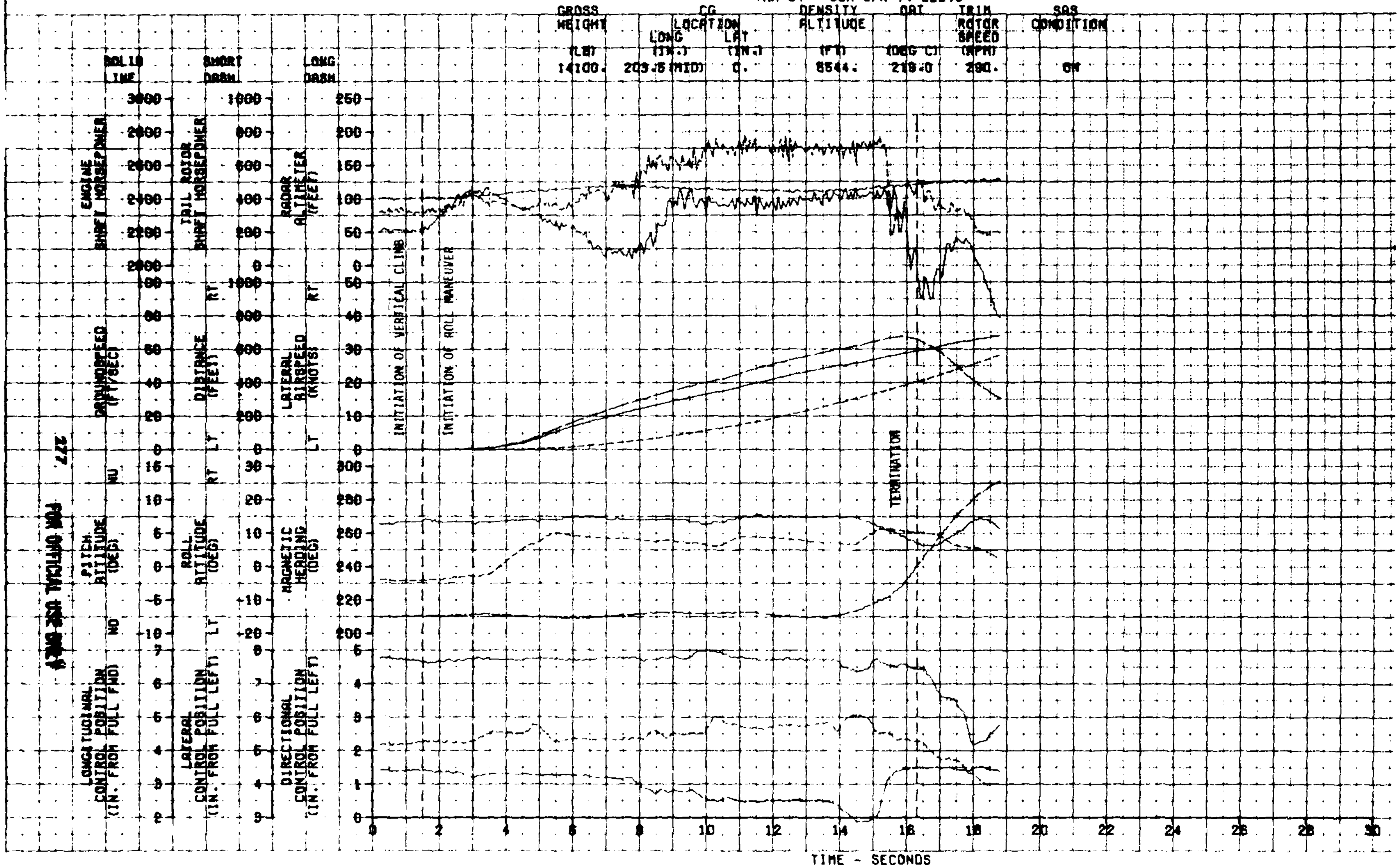


FIGURE 94
LATERAL ACCELERATION
YAH-64 USA S/N 74-22249

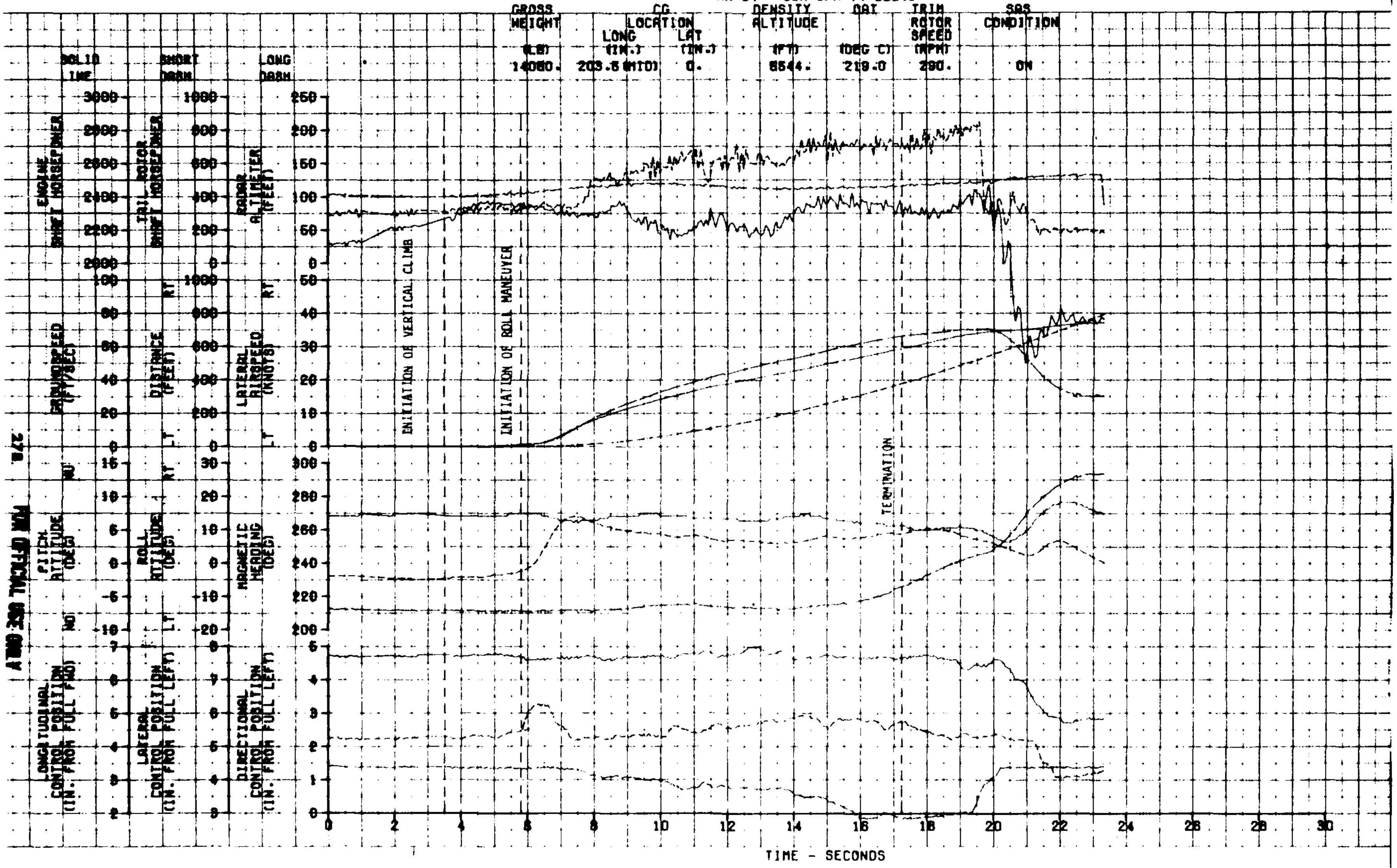


FOR OFFICIAL USE ONLY

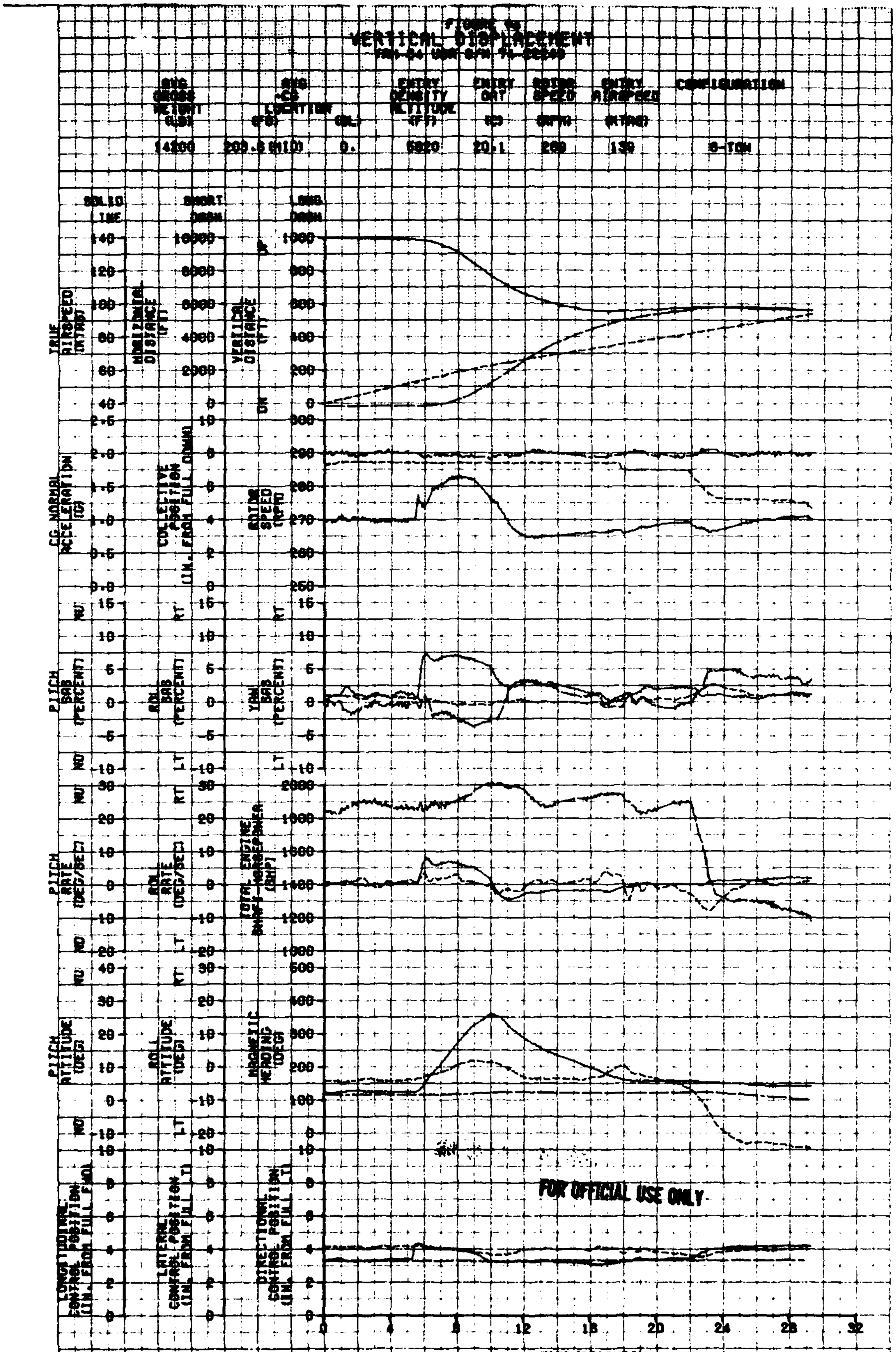
FIGURE 95

LATERAL ACCELERATION

YAH-64 USA S/N 74-22249



FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY

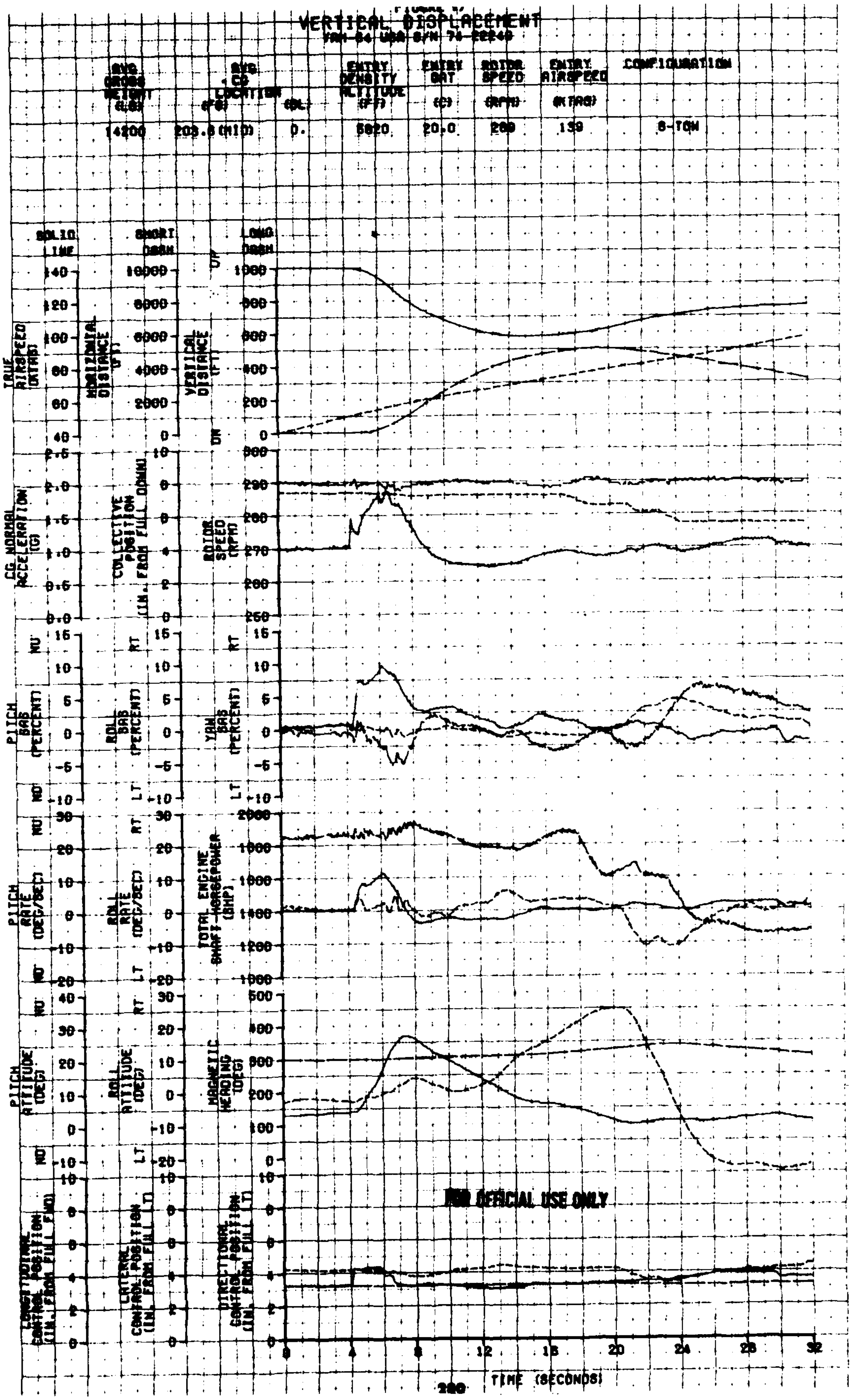
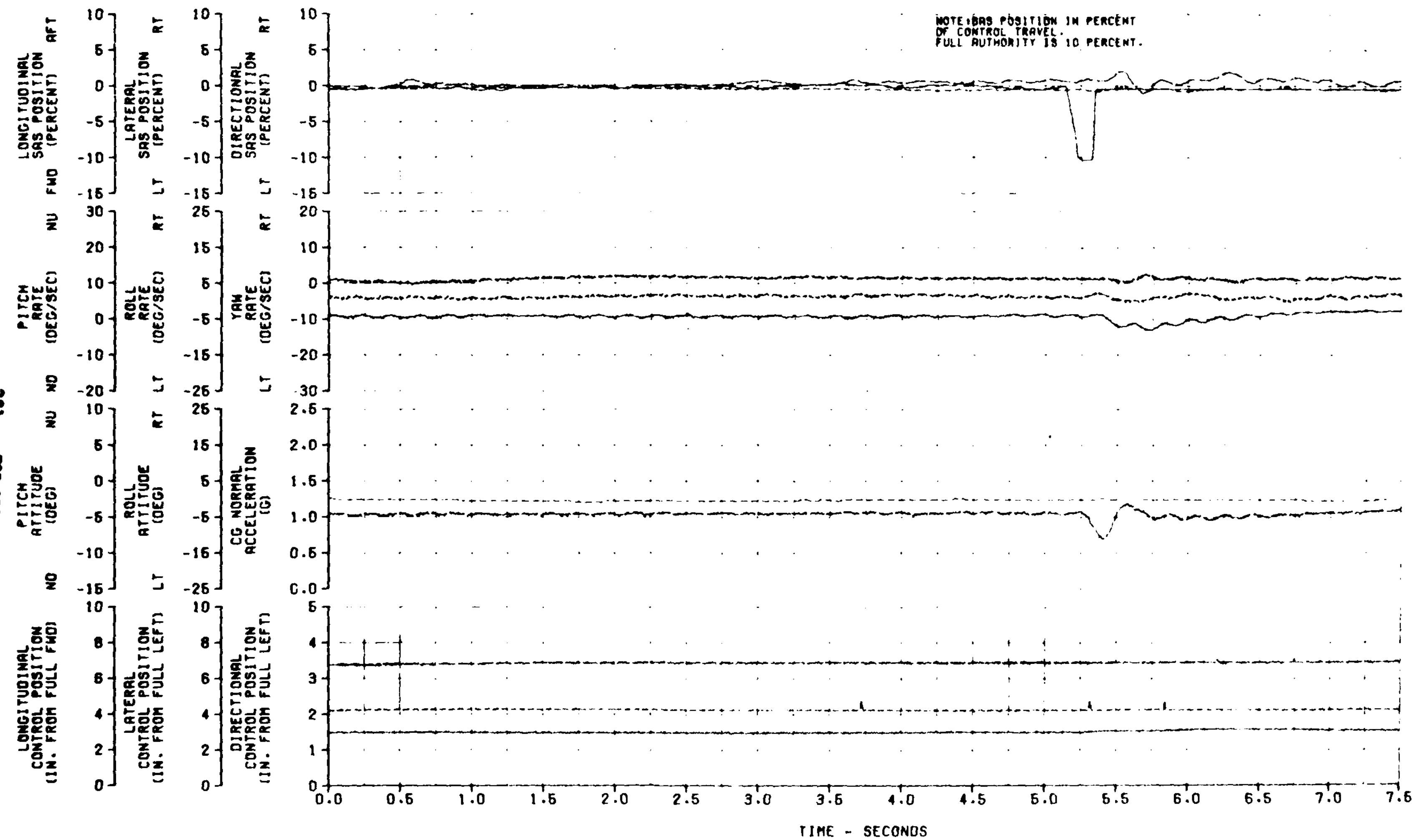

VERTICAL DISPLACEMENT
YAH-64 USA S/N 74-22248
AVG. GROSS WEIGHT (LB)
AVG. CG LOCATION (FS) (BL)
ENTRY DENSITY ALTITUDE (FT)
ENTRY OAT (C)
ROTOR SPEED (RPM)
ENTRY AIRSPEED (KTAS)
CONFIGURATION
14200
203.8 (MID)
0.
5820
20.0
289
139
8-TOW
SOLID LINE
SHORT DASH
LONG DASH
TRUE AIRSPEED (KTAS)
HORIZONTAL DISTANCE (FT)
VERTICAL DISTANCE (FT)
UP
DN
CG NORMAL ACCELERATION (G)
COLLECTIVE POSITION (IN. FROM FULL DOWN)
ROTOR SPEED (RPM)
PITCH SAS (PERCENT)
ROLL SAS (PERCENT)
YAW SAS (PERCENT)
NU
ND
RT
LT
PITCH RATE (DEG/SEC)
ROLL RATE (DEG/SEC)
TOTAL ENGINE SHAFT HORSEPOWER (SHP)
PITCH ATTITUDE (DEG)
ROLL ATTITUDE (DEG)
MAGNETIC HEADING (DEG)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
LATERAL CONTROL POSITION (IN. FROM FULL LT)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)
FOR OFFICIAL USE ONLY
TIME (SECONDS)

FIGURE 98

LONGITUDINAL SAS HARDOVER FAILURE

YAH-64 USA S/N 74-22248

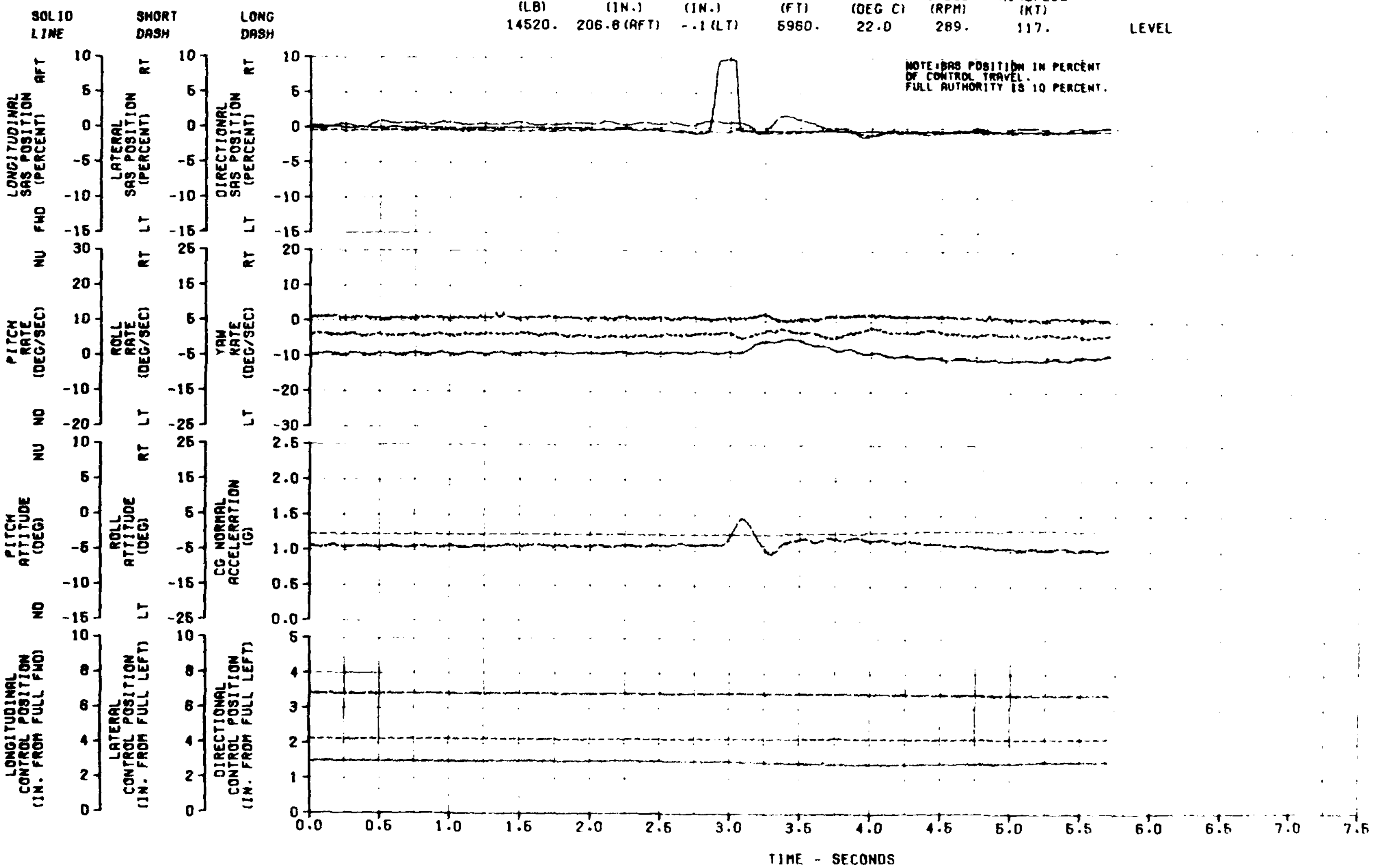
GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14520.	206.8 (AFT)	-.1 (LT)	5960.	22.0	289.	117.	LEVEL



FLIGHT 33 RUN 24 120 KIAS FWD LONGITUDINAL HARDOVER

FIGURE 99
LONGITUDINAL SAS HARDOVER FAILURE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14520.	206.8 (AFT)	-.1 (LT)	5960.	22.0	289.	117.	LEVEL



FLIGHT 33 RUN 23 120 KIAS AFT LONGITUDINAL HARDOVER

FIGURE 100
LATERAL SAS HARDOVER FAILURE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14520.	206.8(AFT)	-.1(LT)	5960.	22.0	289.	117.	LEVEL

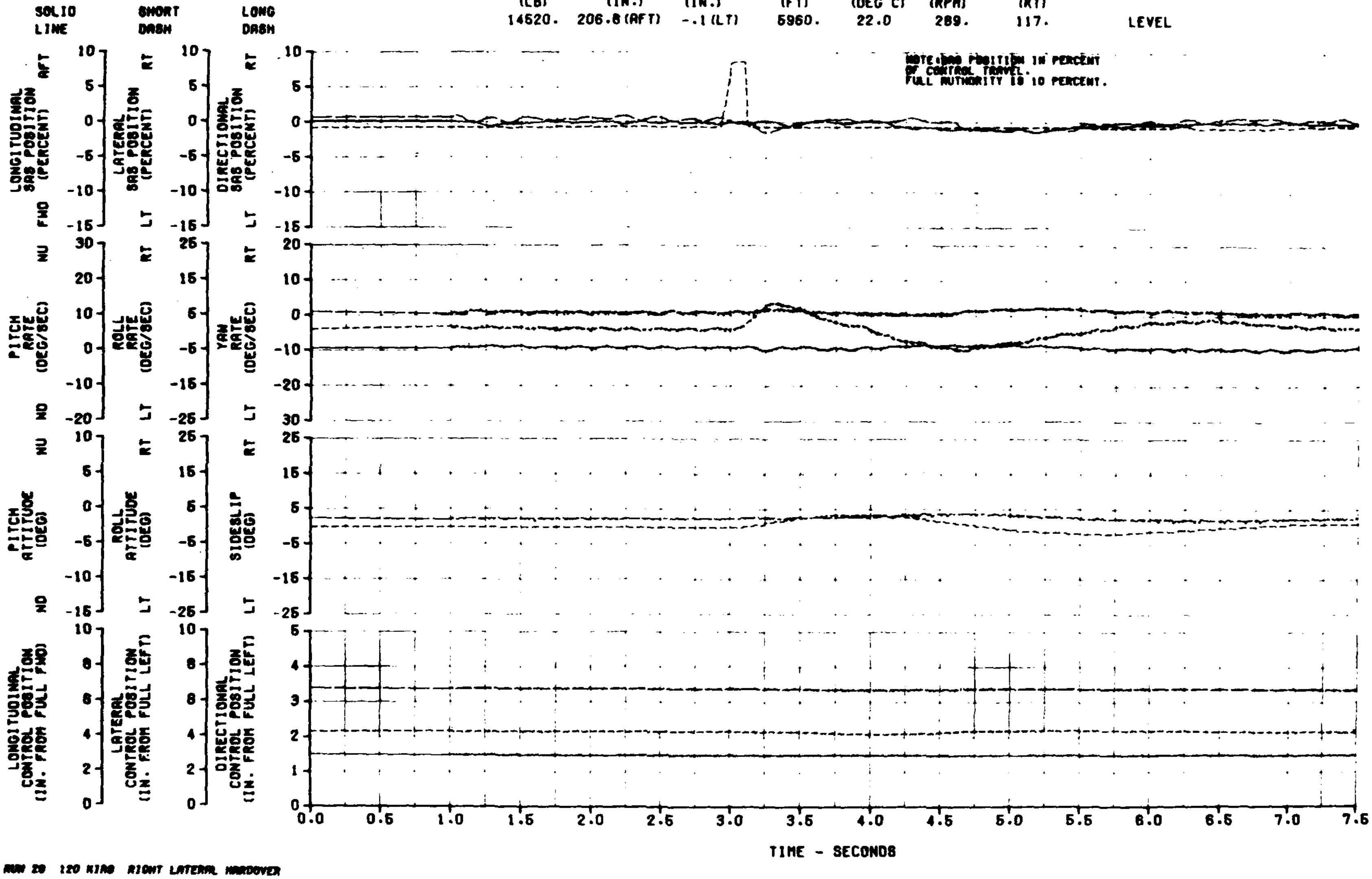
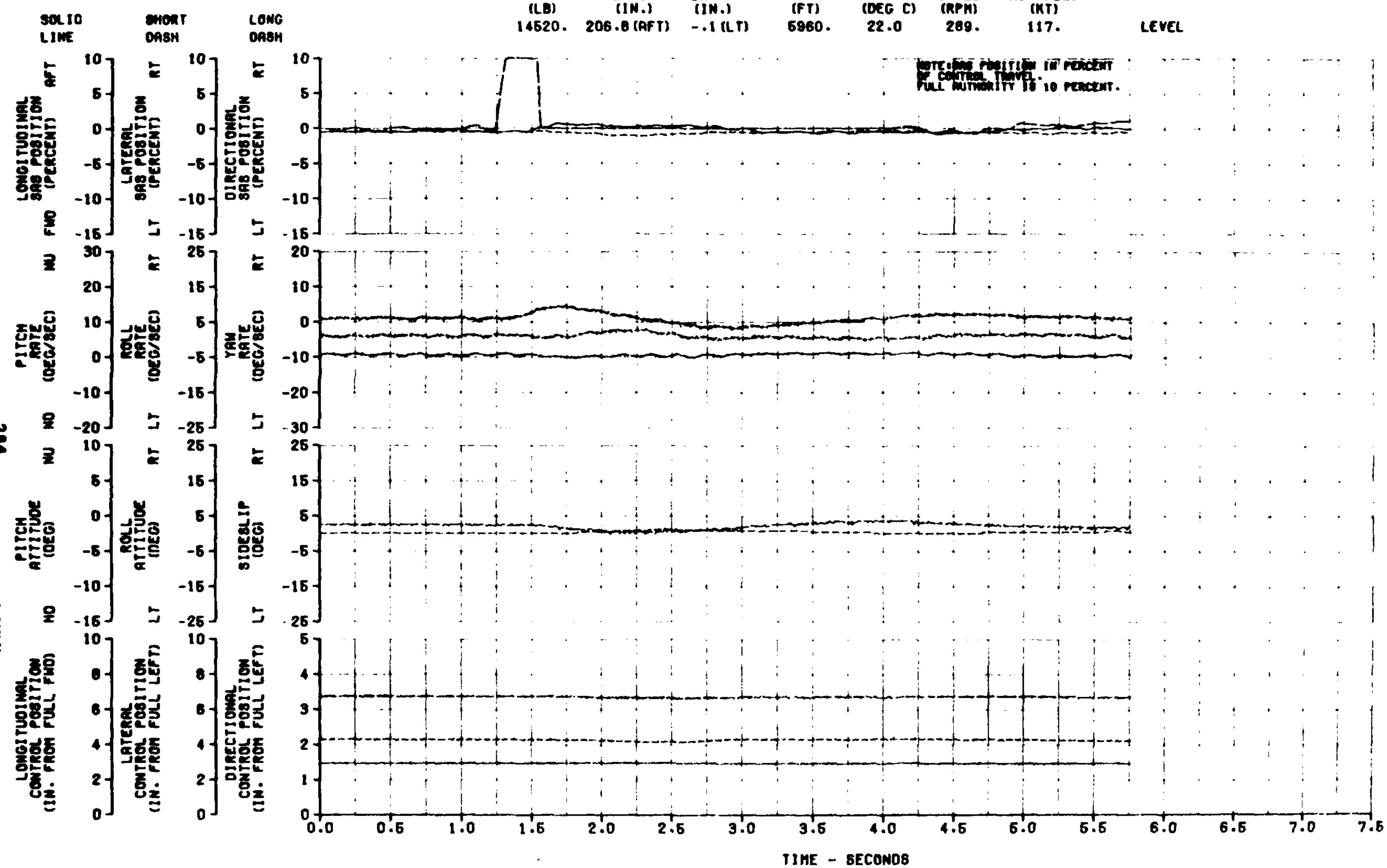


FIGURE 101

DIRECTIONAL SAS HARDOVER FAILURE

YAH-64 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14520.	206.8 (AFT)	-.1 (LT)	5960.	22.0	289.	117.	LEVEL



FLIGHT 33 RUN 28 120 KIAS RIGHT DIRECTIONAL HARDOVER

ENGINE FAILURE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG. (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14500.	206.9 (AFT)	-.1 (LT)	7700.	18.0	289.	80.	LEVEL FLIGHT

NOTES: 1) SIMULATED NO. 1 ENGINE FAILURE FROM DUAL ENGINE FLIGHT

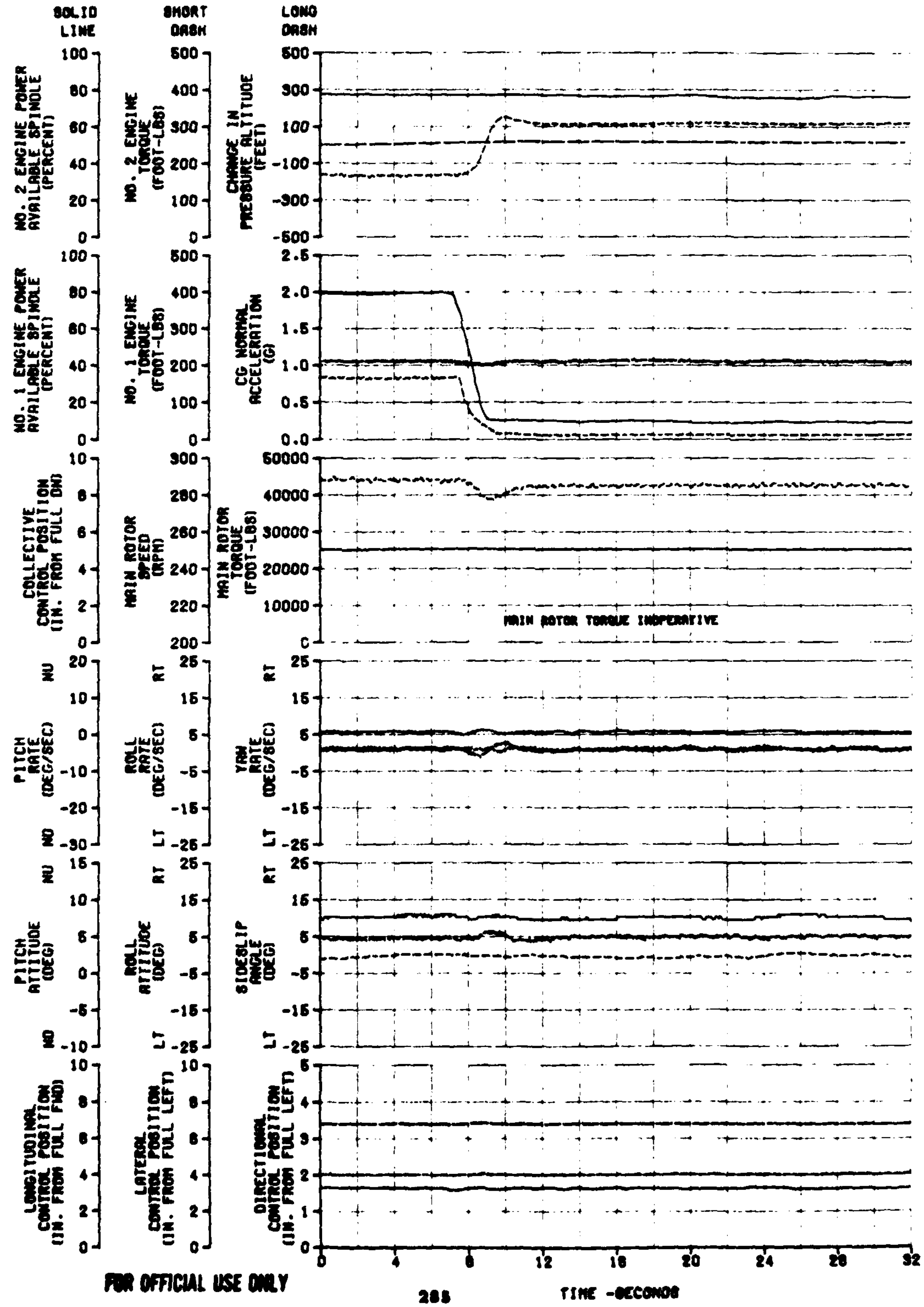


FIGURE 103

ENGINE FAILURE

YAH-64 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14300.	206.8 (AFT)	-.1 (LT)	5840.	24.0	288.	79.	LEVEL FLIGHT

NOTES: 1) SIMULATED NO. 1 ENGINE FAILURE FROM SINGLE ENGINE FLIGHT

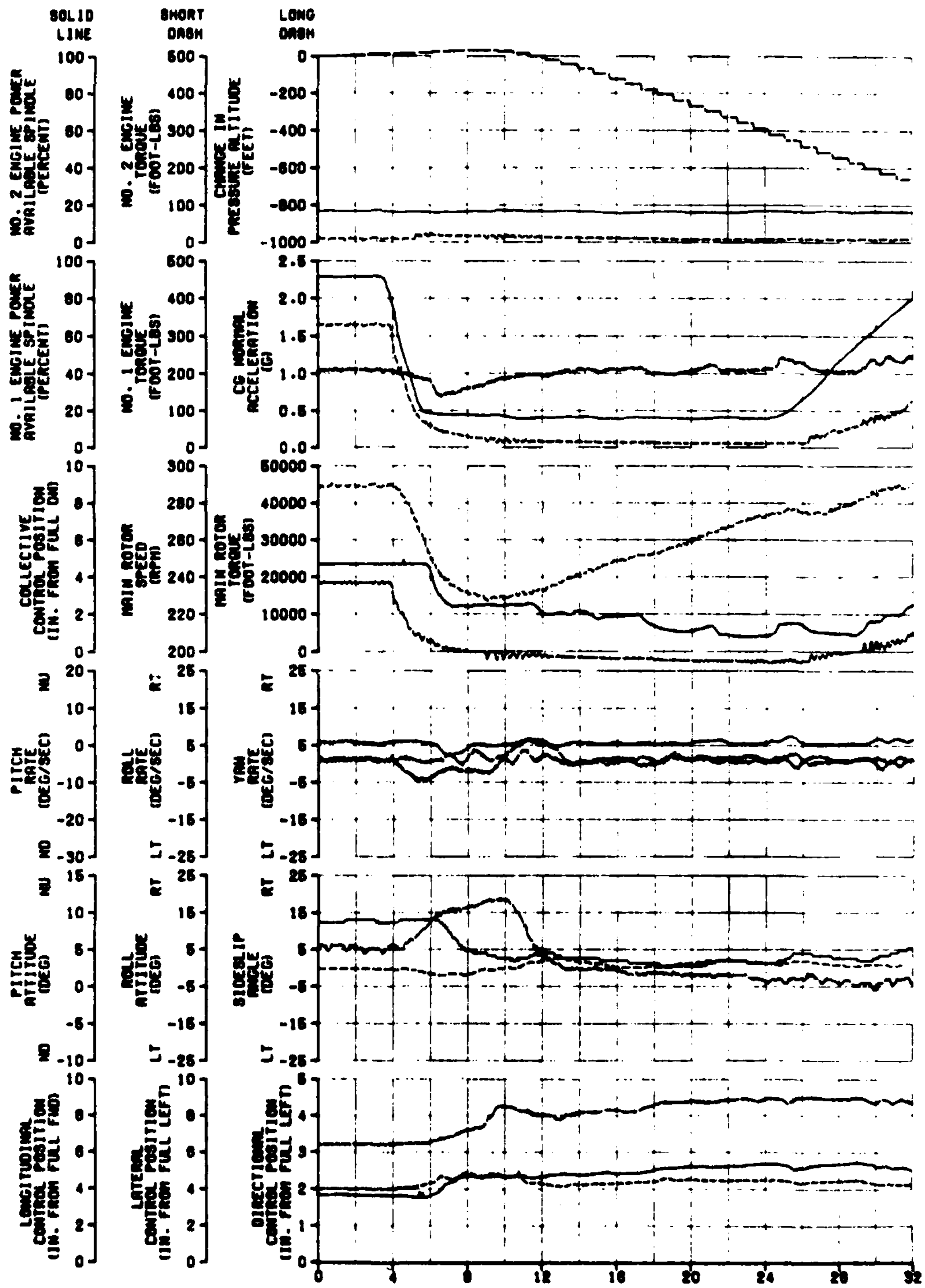


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FIGURE 104
ENGINE FAILURE
YAH-64 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14420.	206.7(AFT)	-.1(LT)	6100.	24.0	289.	76.	2000 FPM CLIMB

NOTES: 1) SIMULATED NO. 1 ENGINE FAILURE FROM DUAL ENGINE FLIGHT

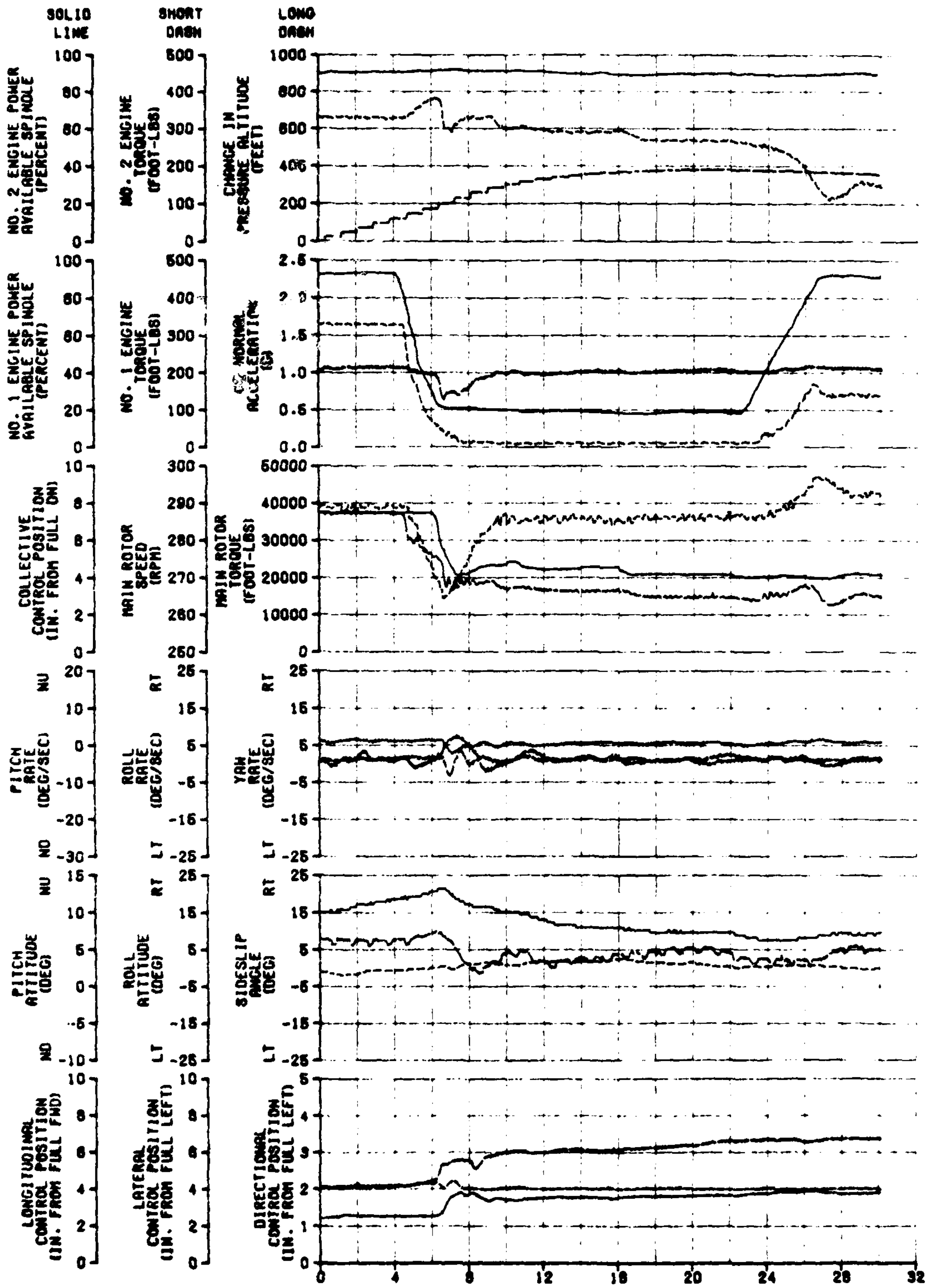

FIGURE 105
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

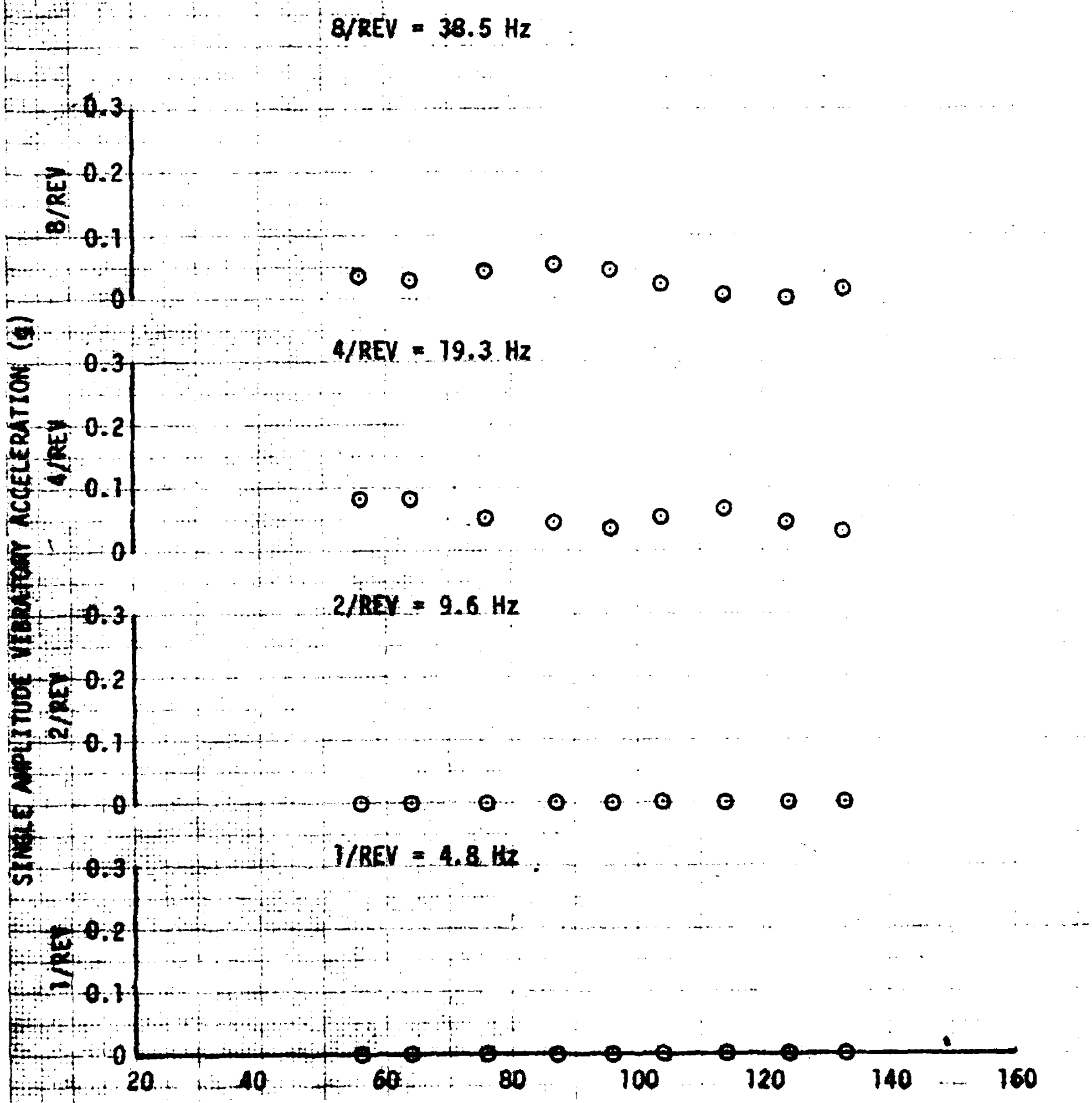

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FIGURE 106
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT SEAT LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

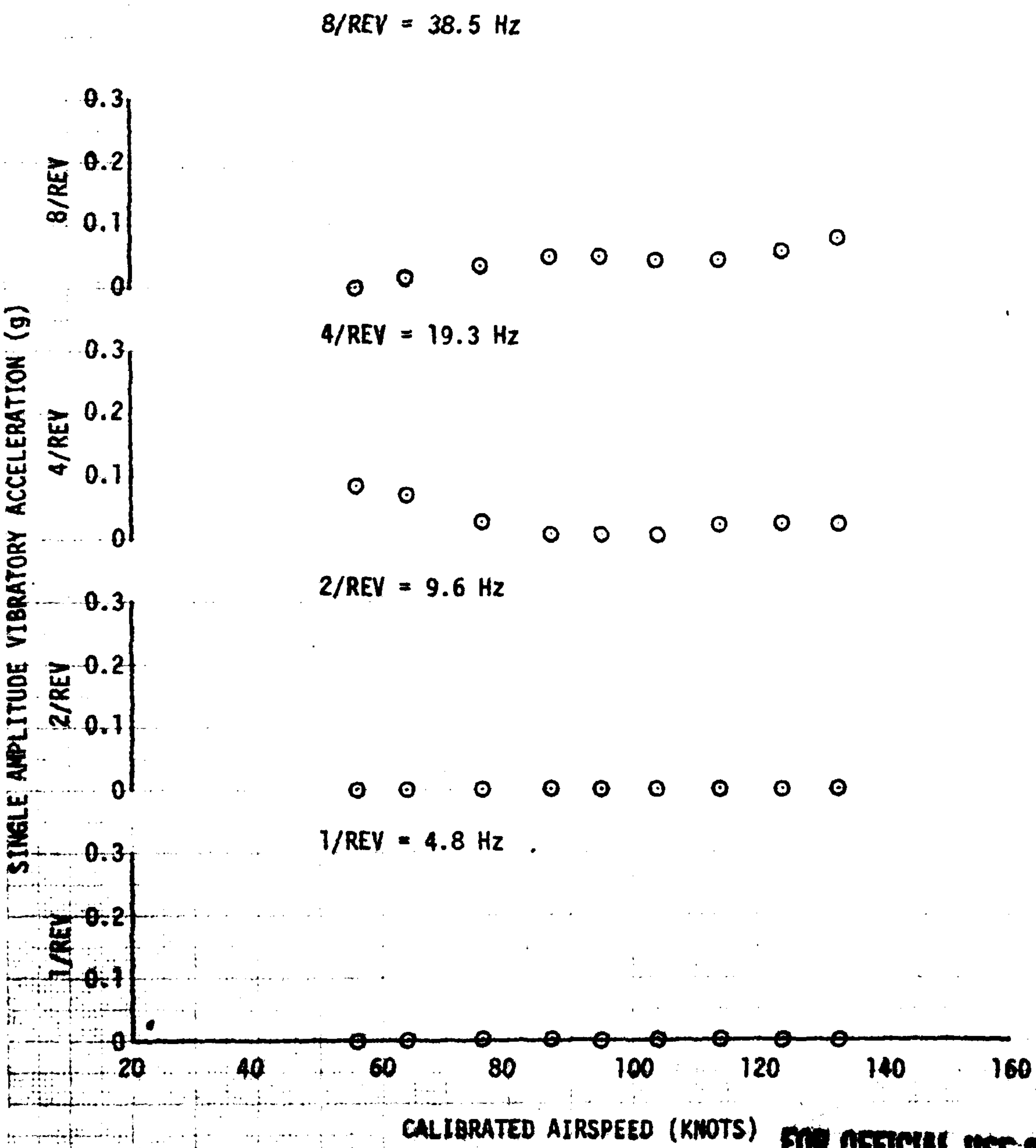

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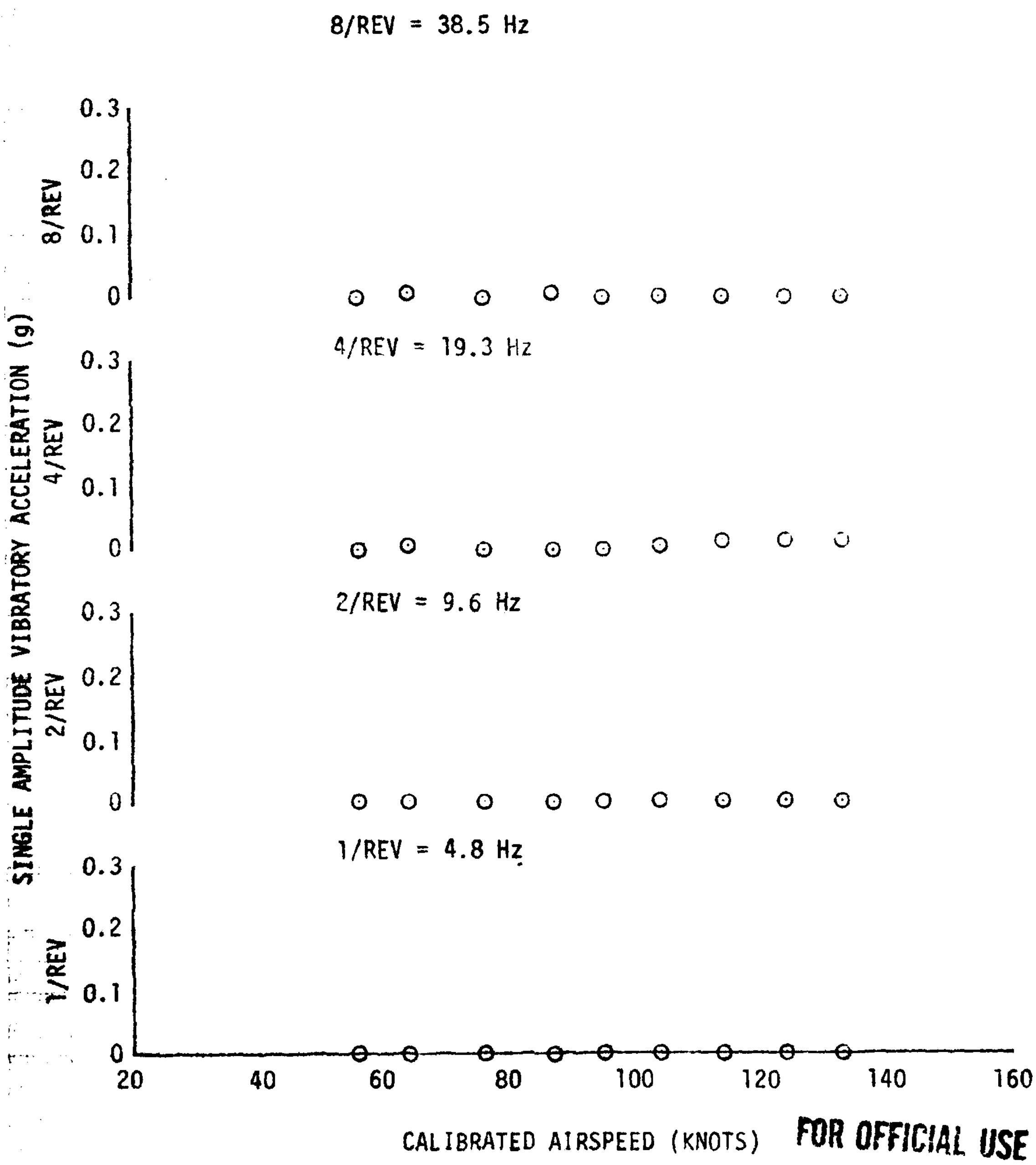
FIGURE 107
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT SEAT LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW



FOR OFFICIAL USE ONLY

FIGURE 108
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT CYCLIC CONTROL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

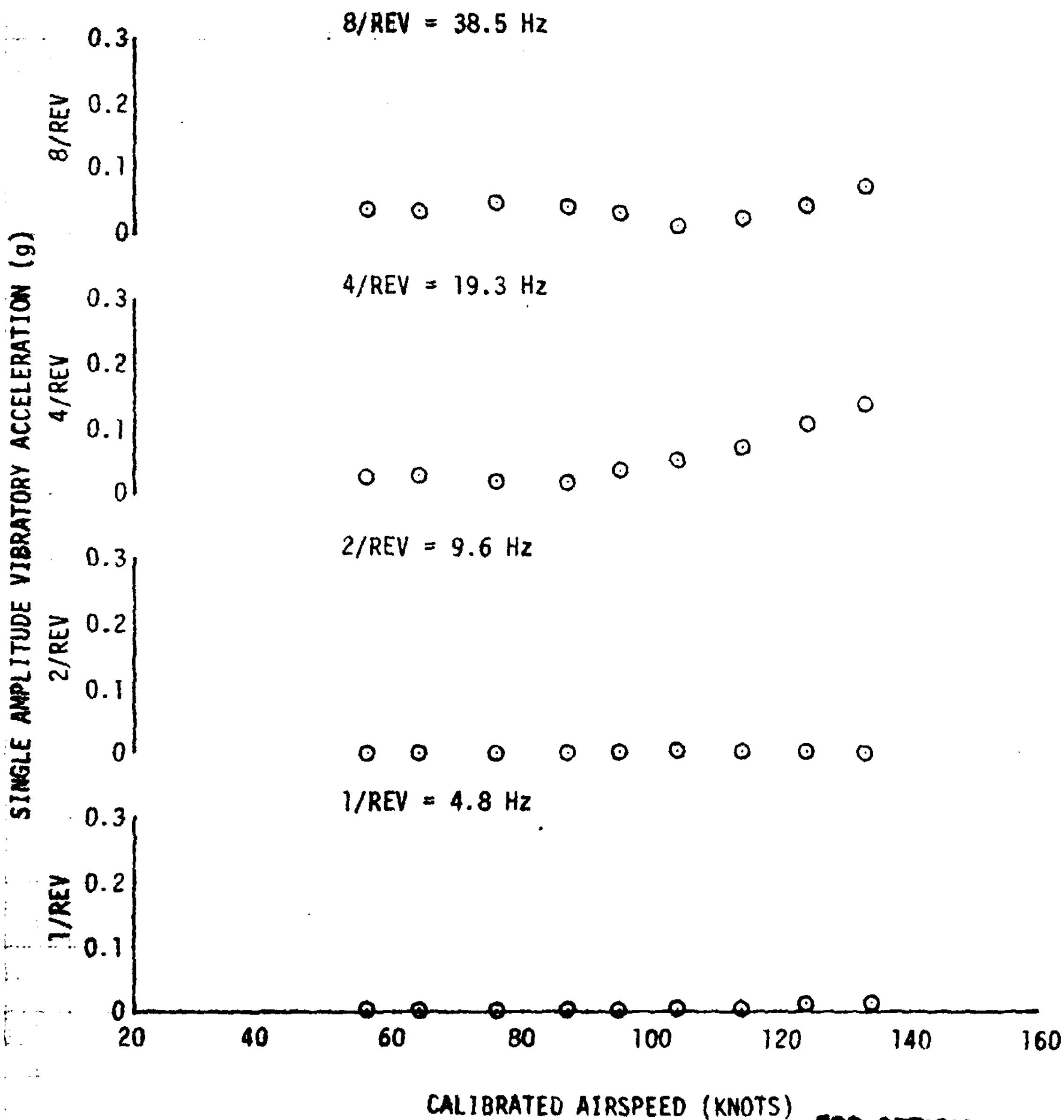


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FIGURE 109
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT CYCLIC CONTROL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.[illegible](MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

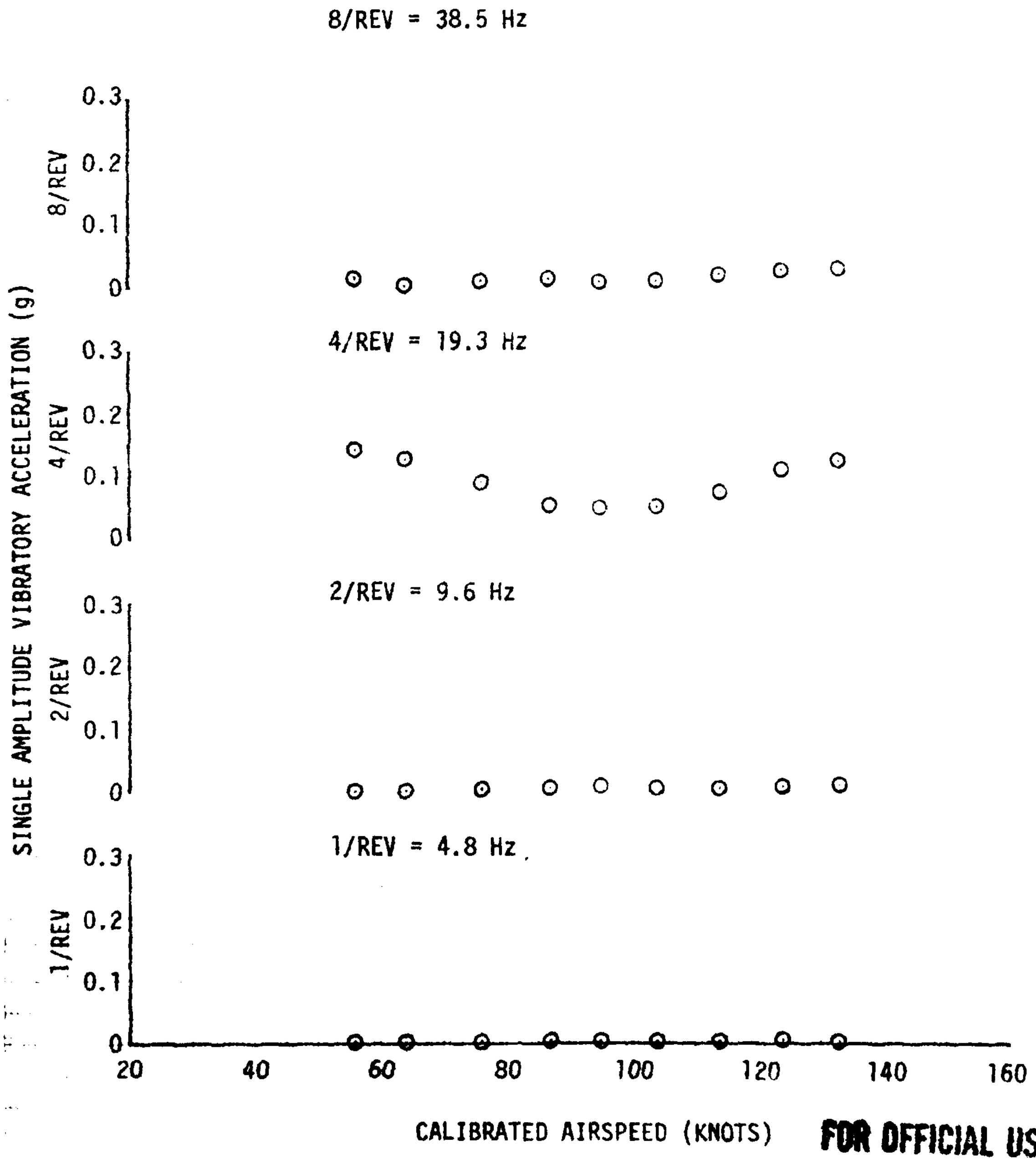

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FIGURE 110
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT CYCLIC CONTROL LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

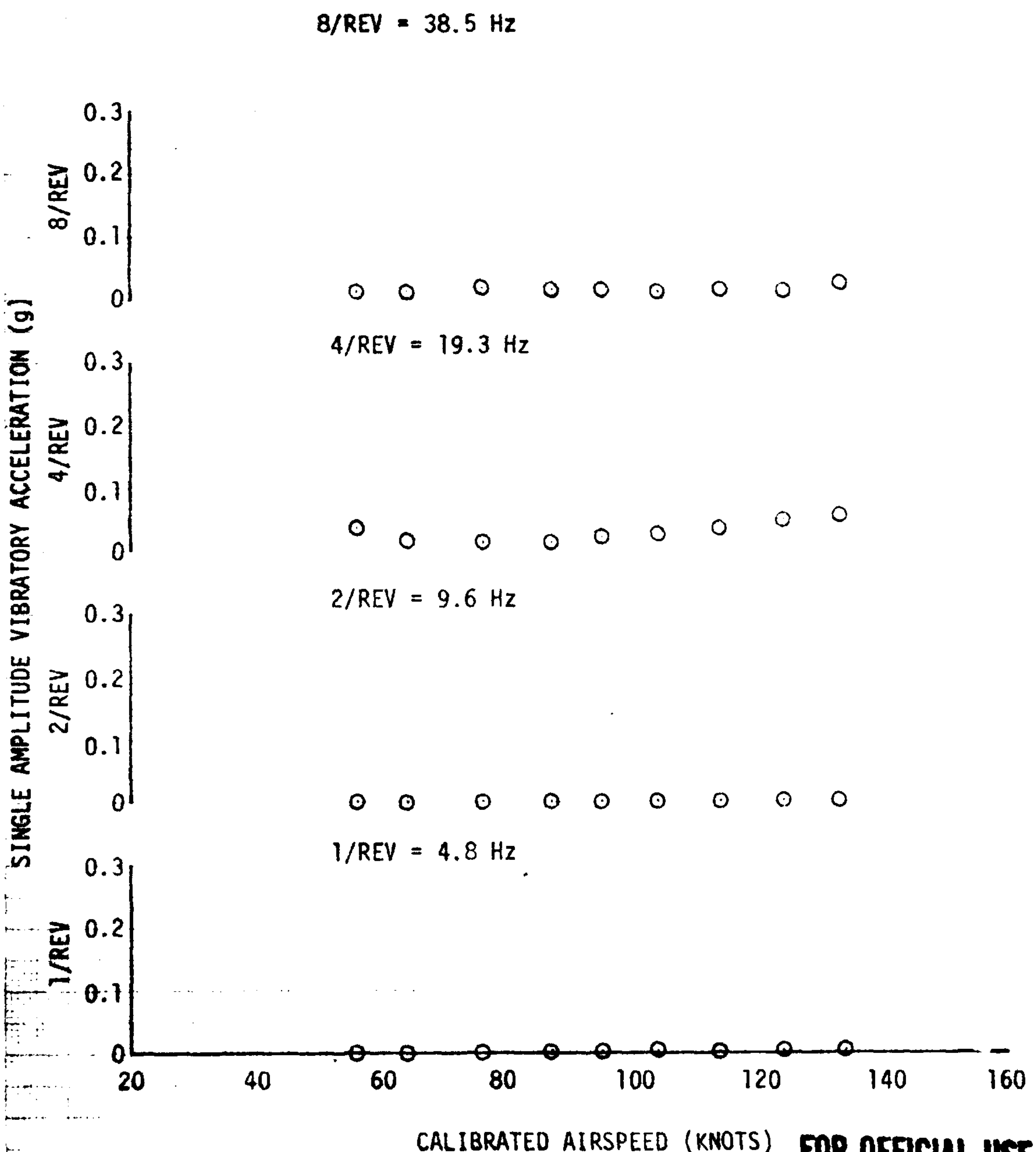

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FIGURE 111
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT COLLECTIVE CONTROL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

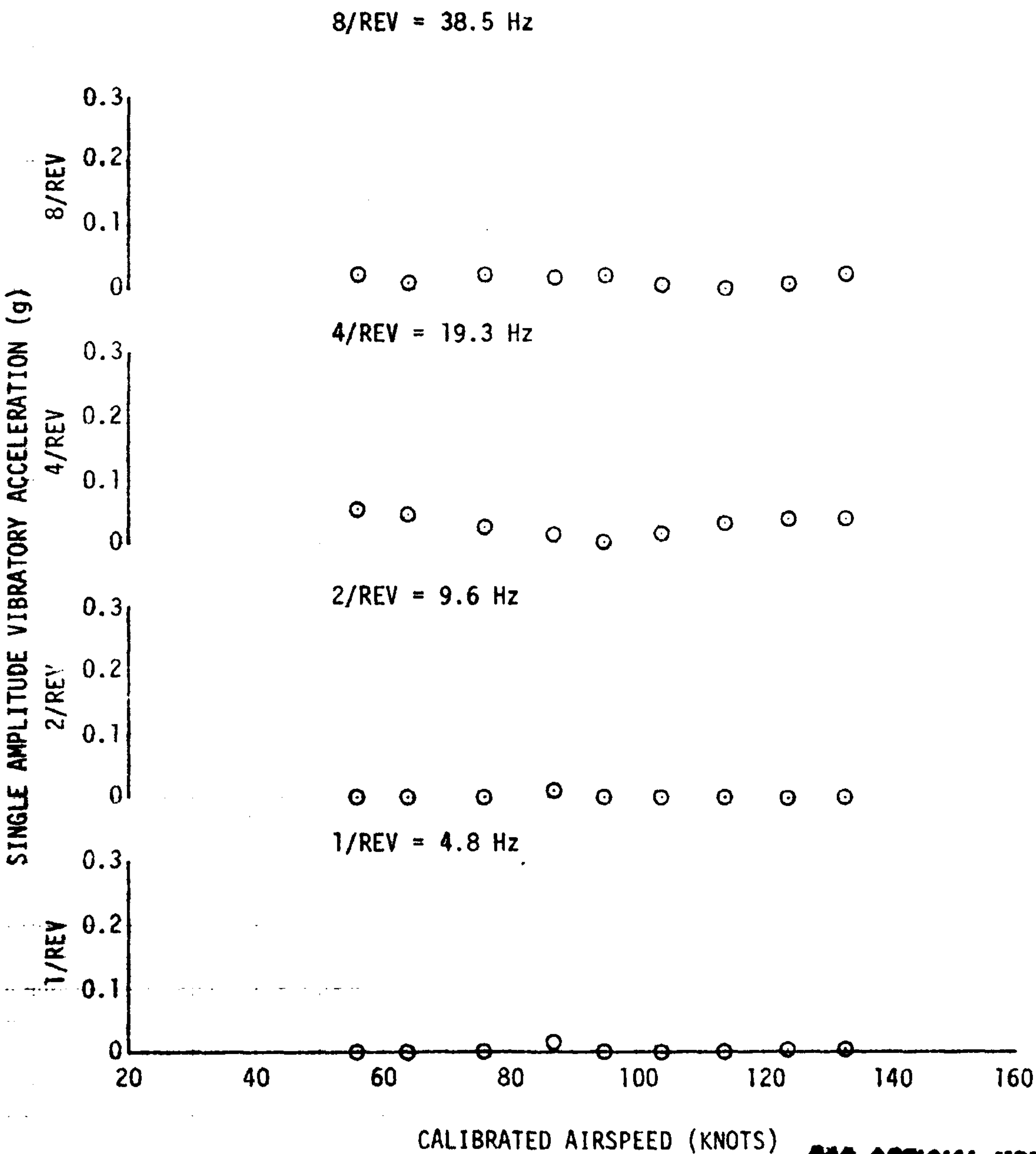

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FIGURE 112
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT COLLECTIVE CONTROL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

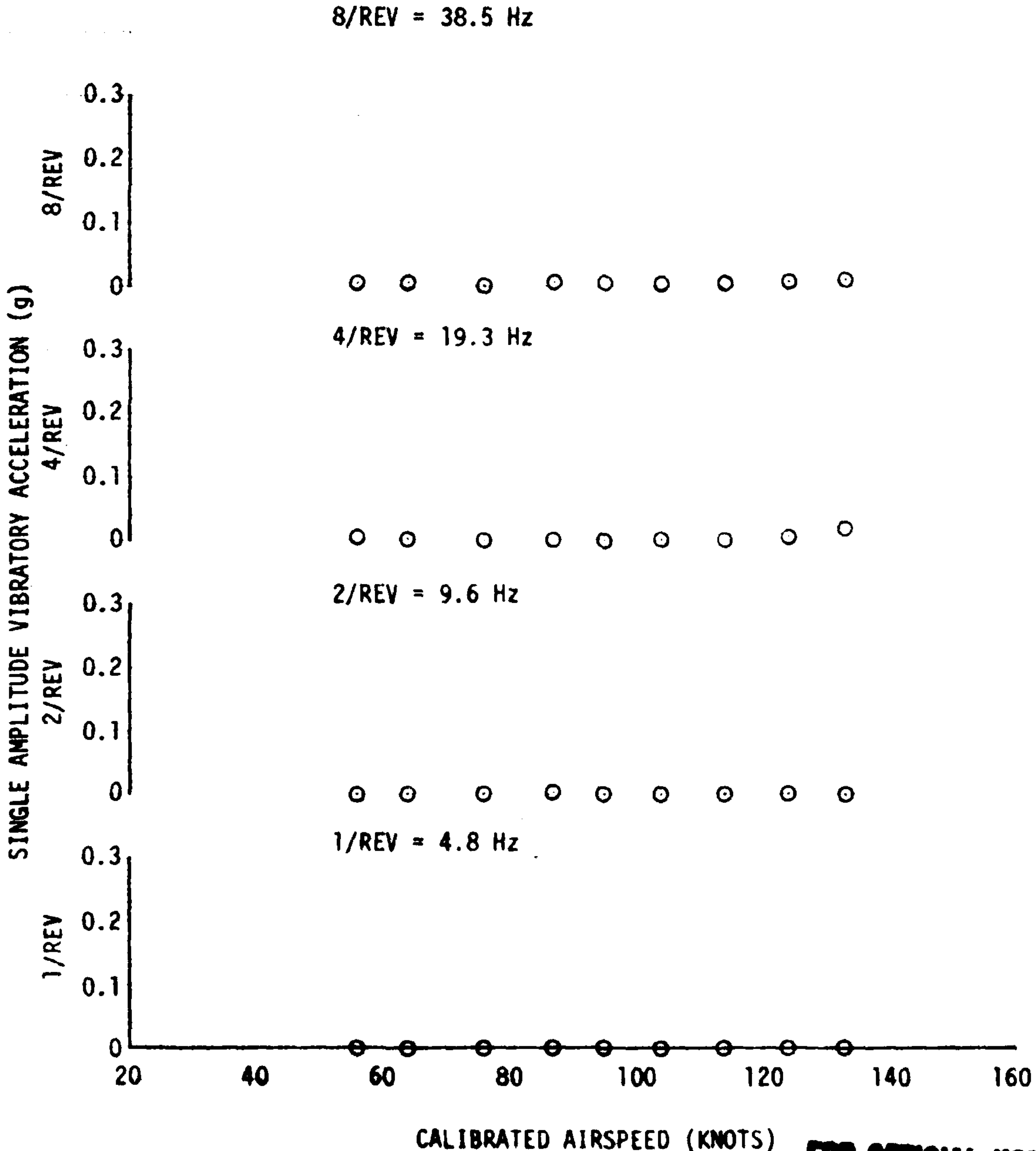

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FIGURE 113
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT HEEL REST VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

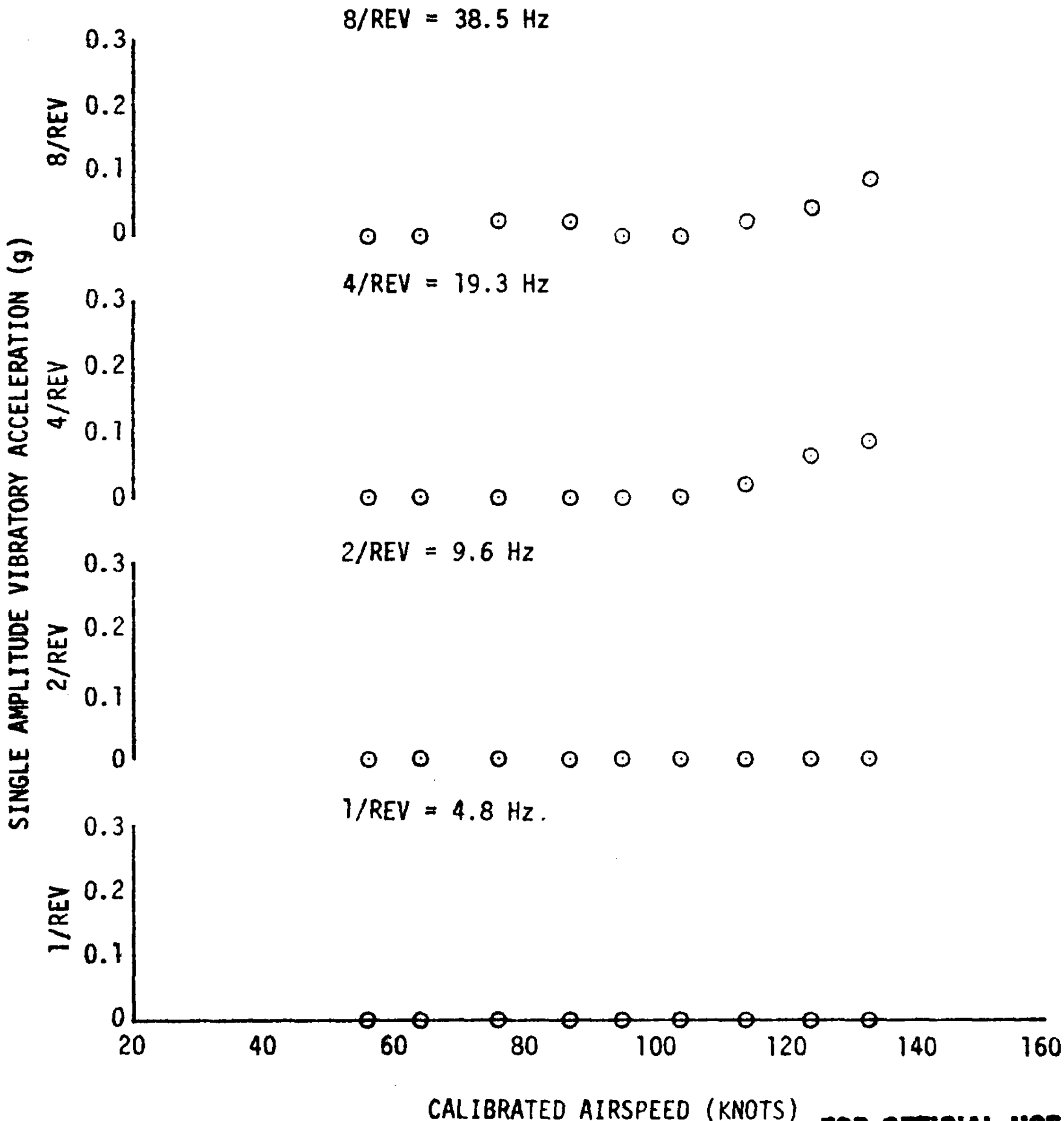

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FIGURE 114
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT HEEL REST LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

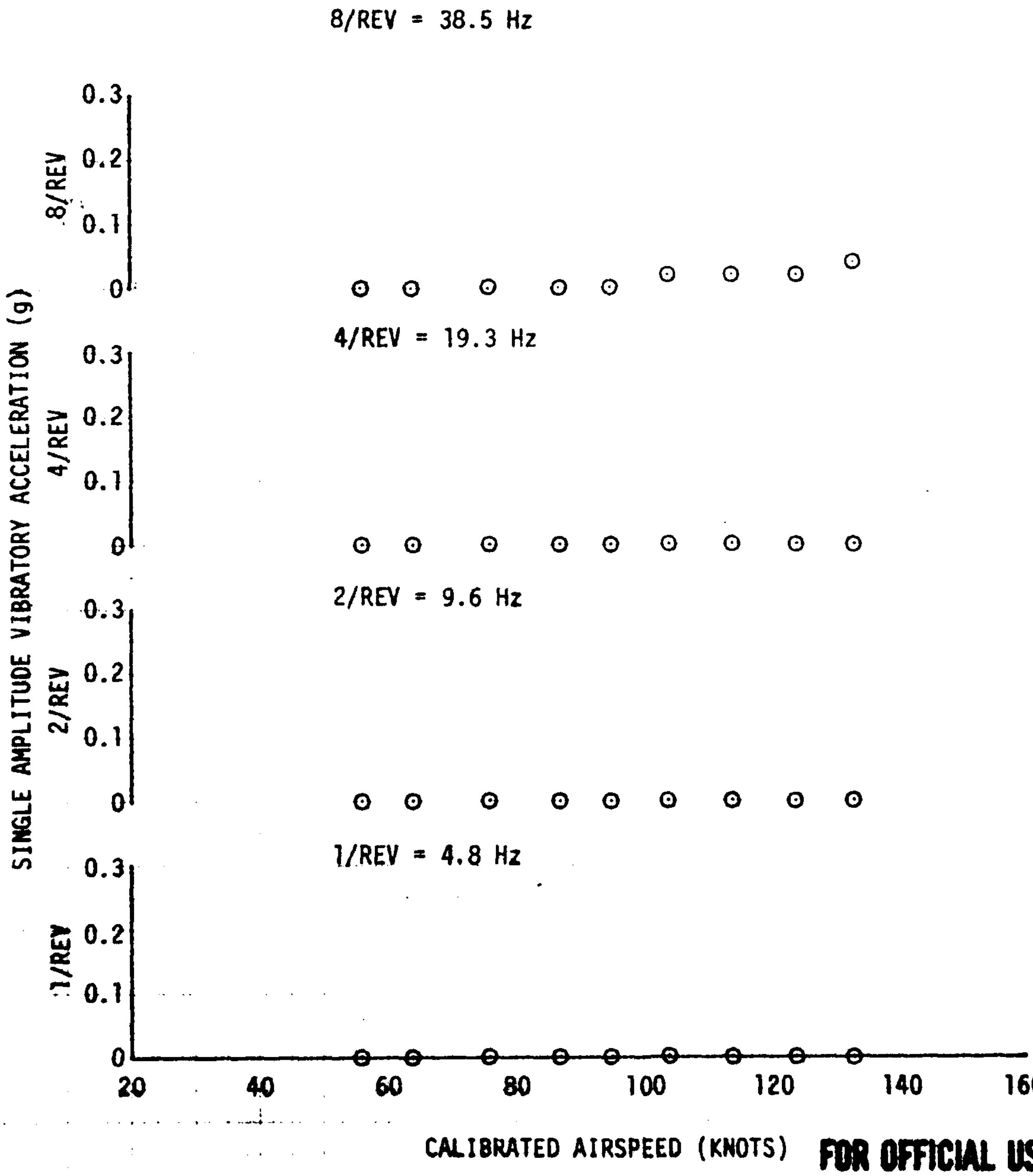

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FIGURE 115
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT INSTRUMENT PANEL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8+TOW

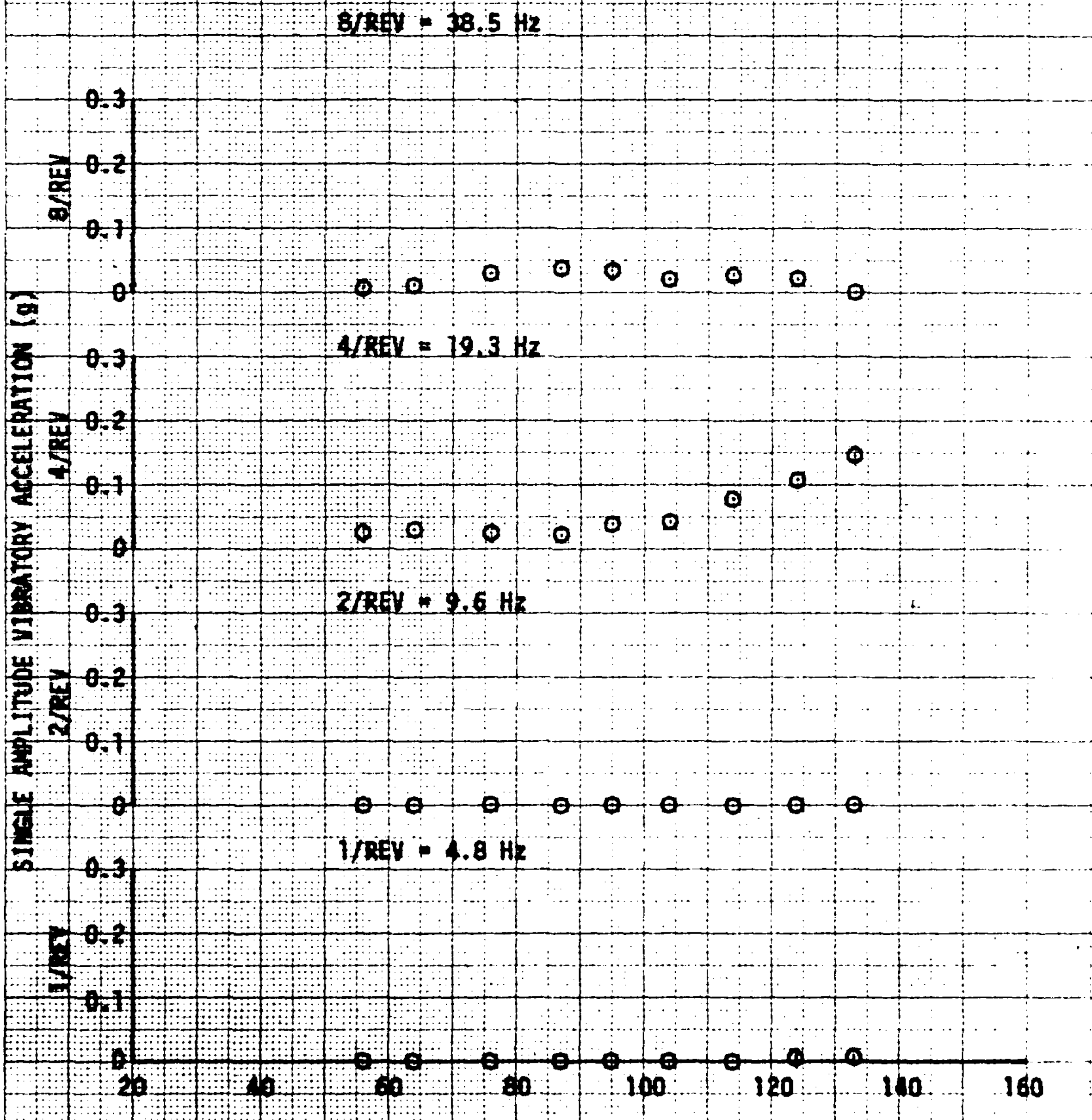

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FIGURE 116
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT INSTRUMENT PANEL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

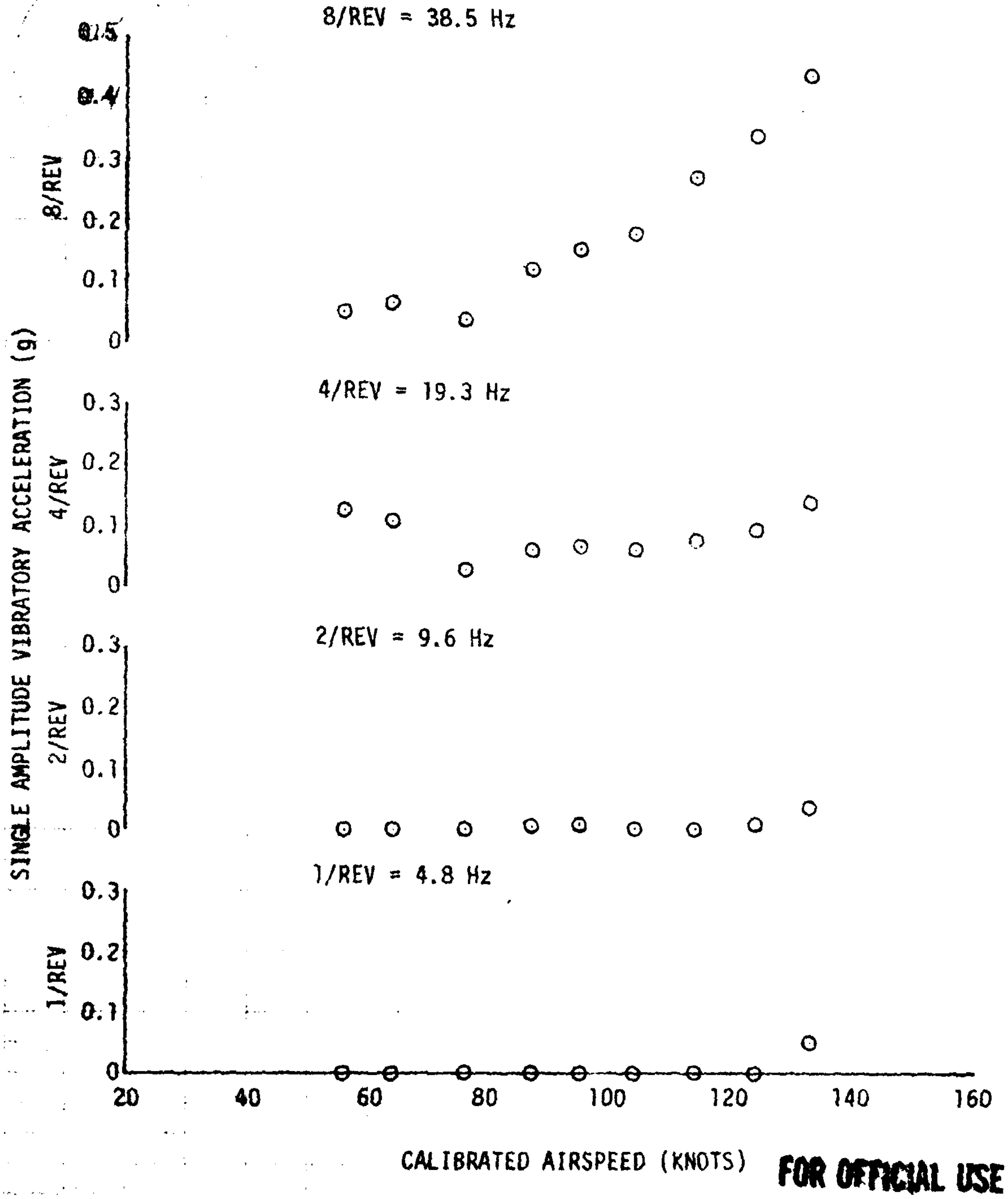

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FIGURE 117
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
PILOT INSTRUMENT PANEL LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

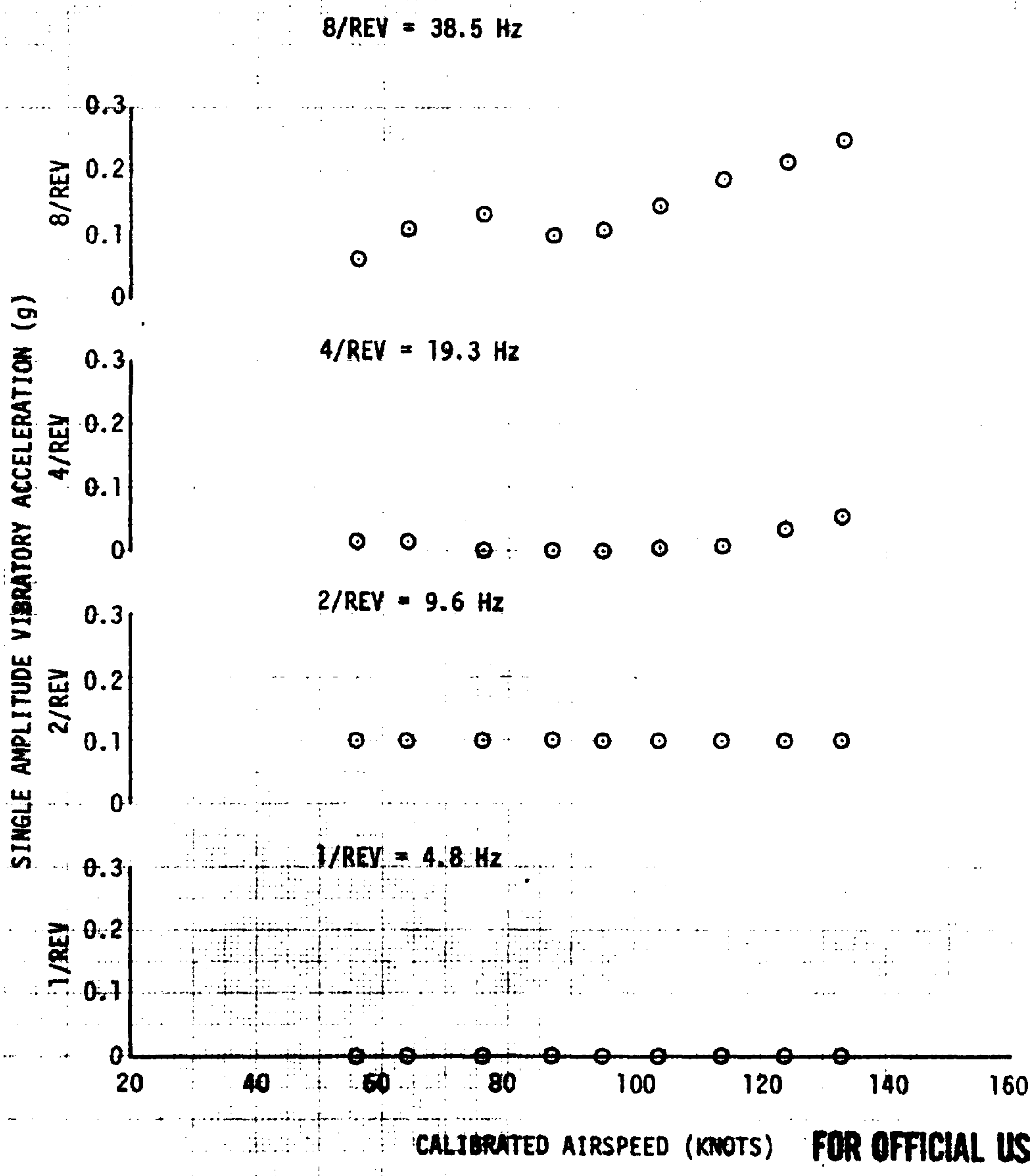

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FIGURE 118
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

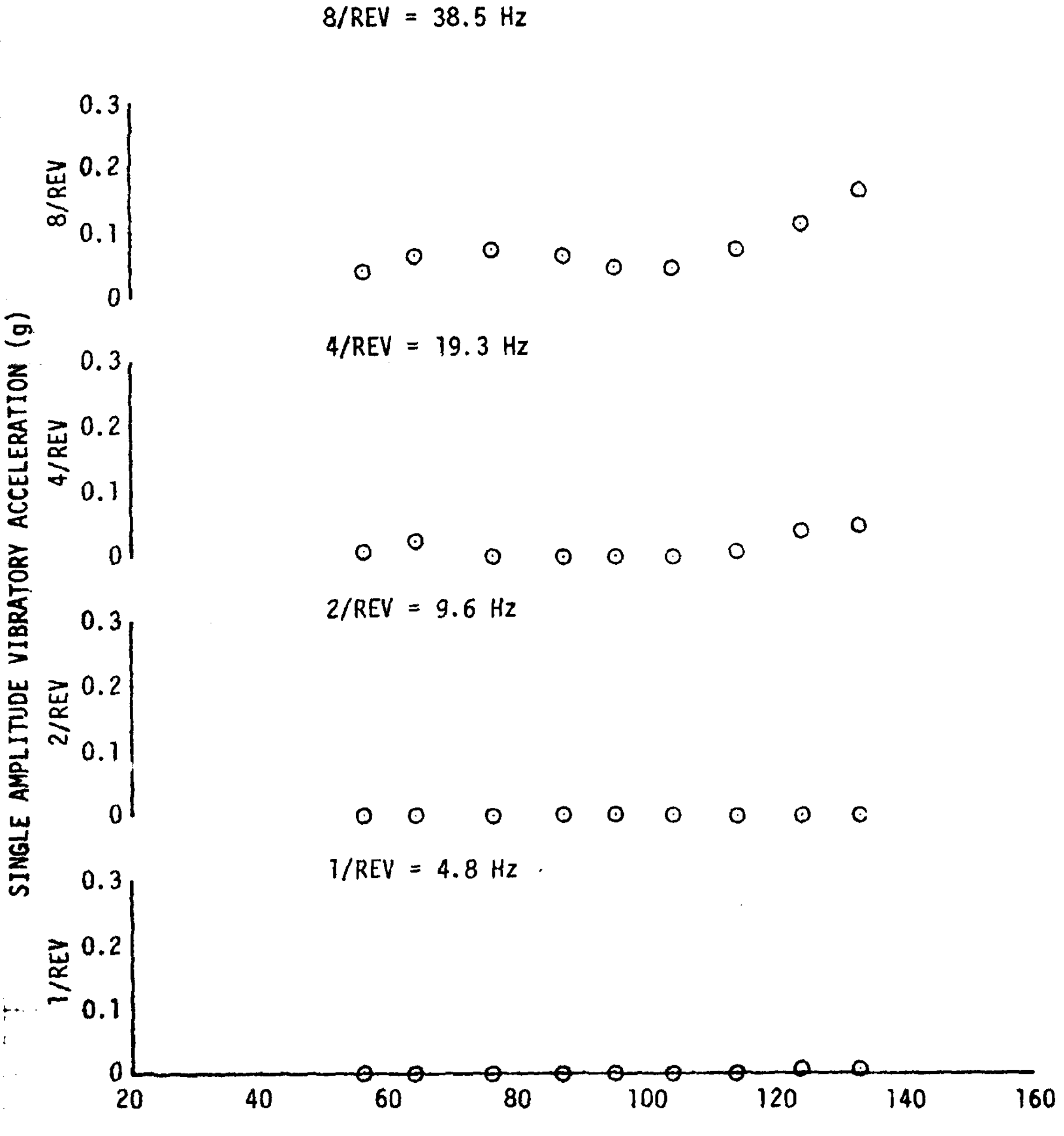

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FIGURE 119
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT SEAT LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

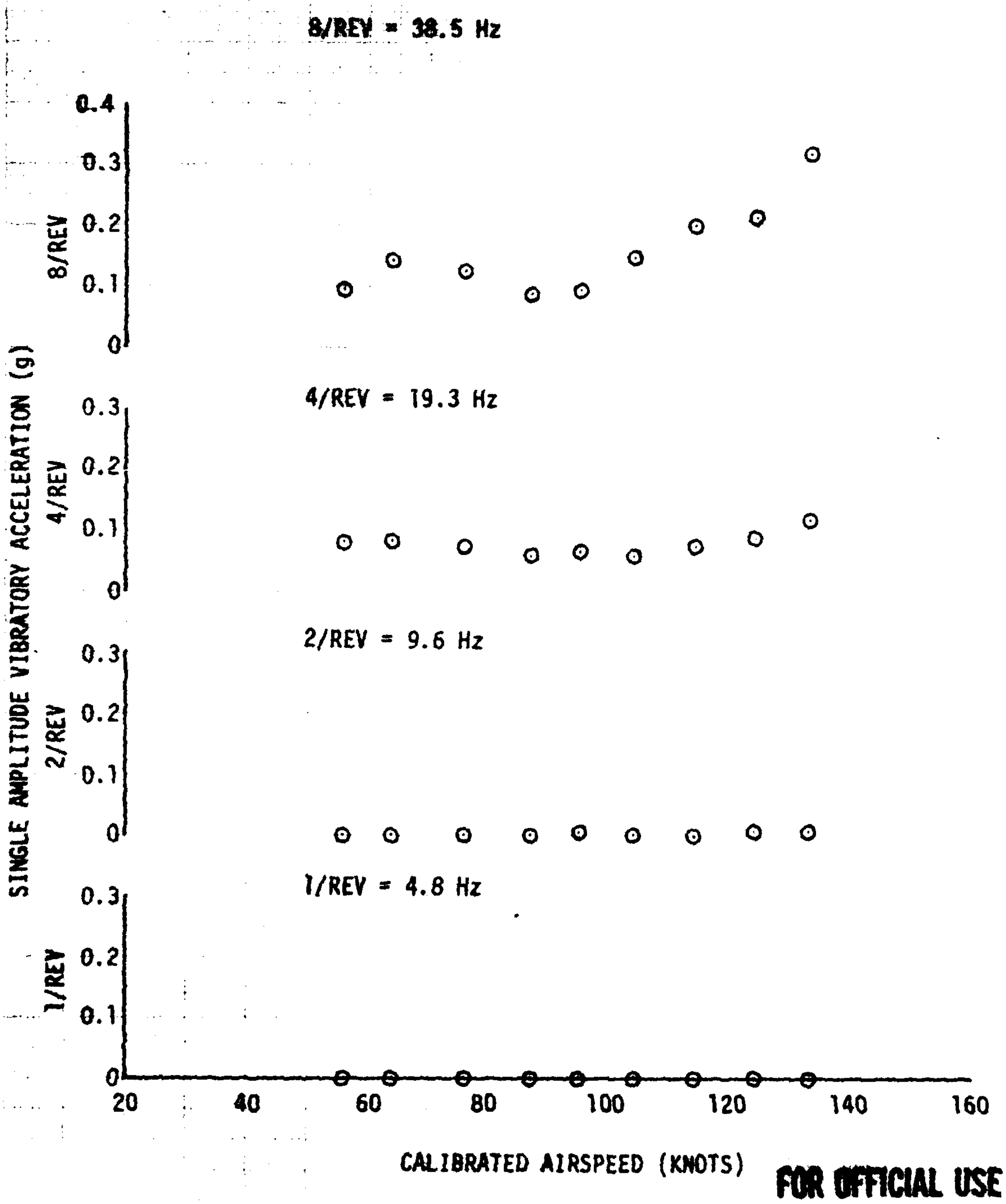

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FIGURE 120
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT SEAT LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

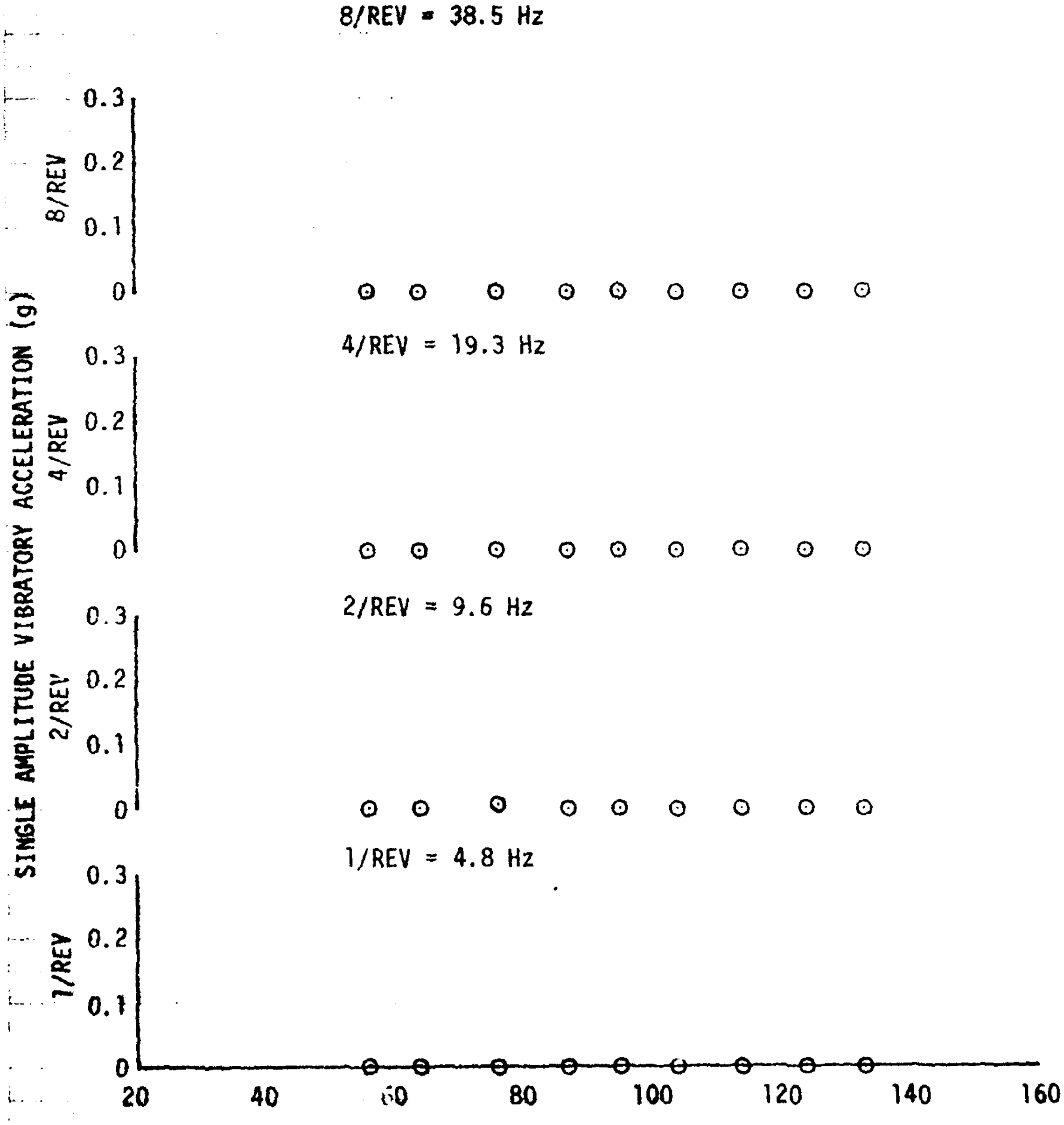

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FIGURE 121
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT INSTRUMENT PANEL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

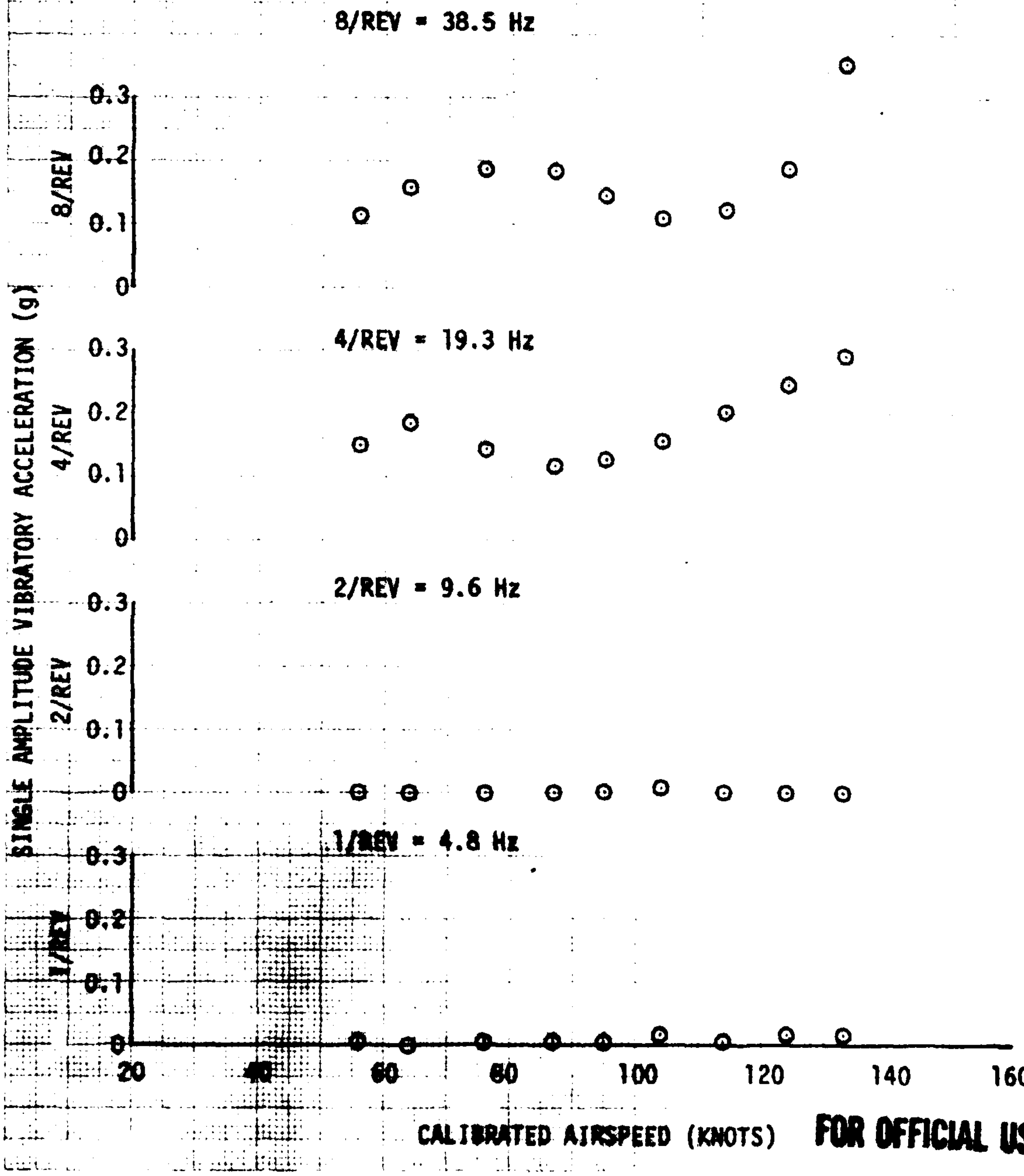

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FIGURE 122
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT INSTRUMENT PANEL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

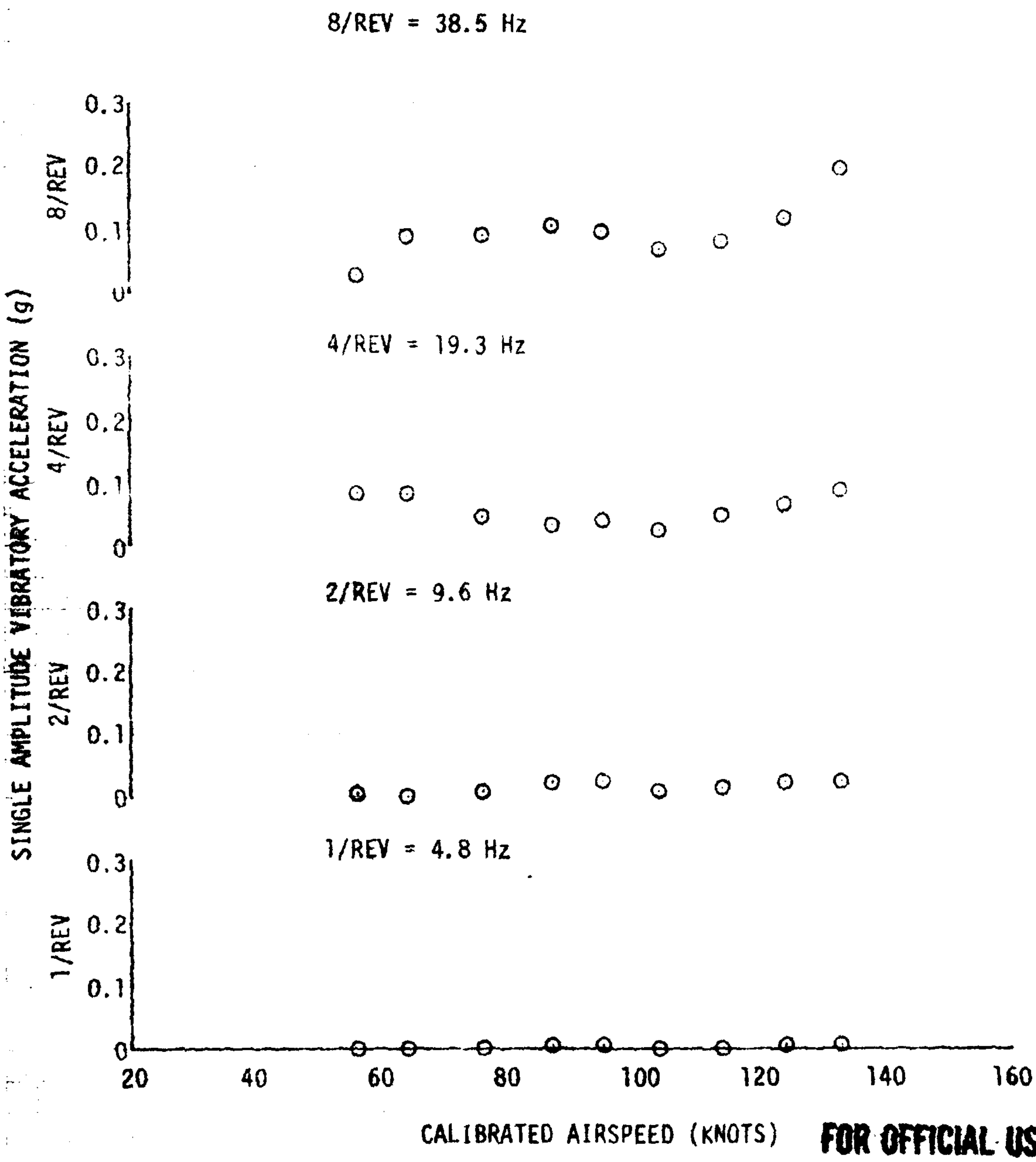

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FIGURE 123
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT INSTRUMENT PANEL LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13680	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

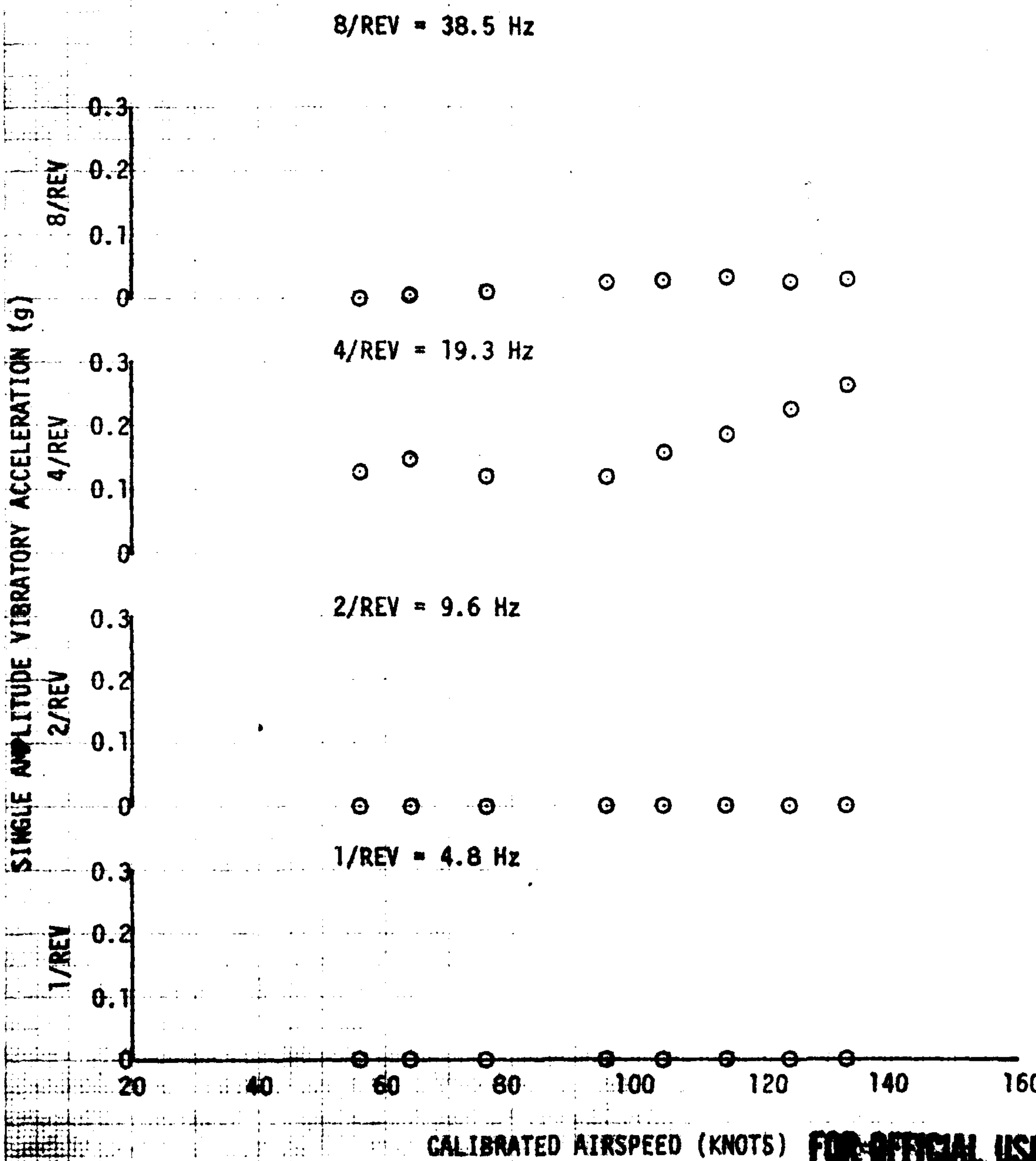

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FIGURE 124
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
AIRCRAFT CG VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

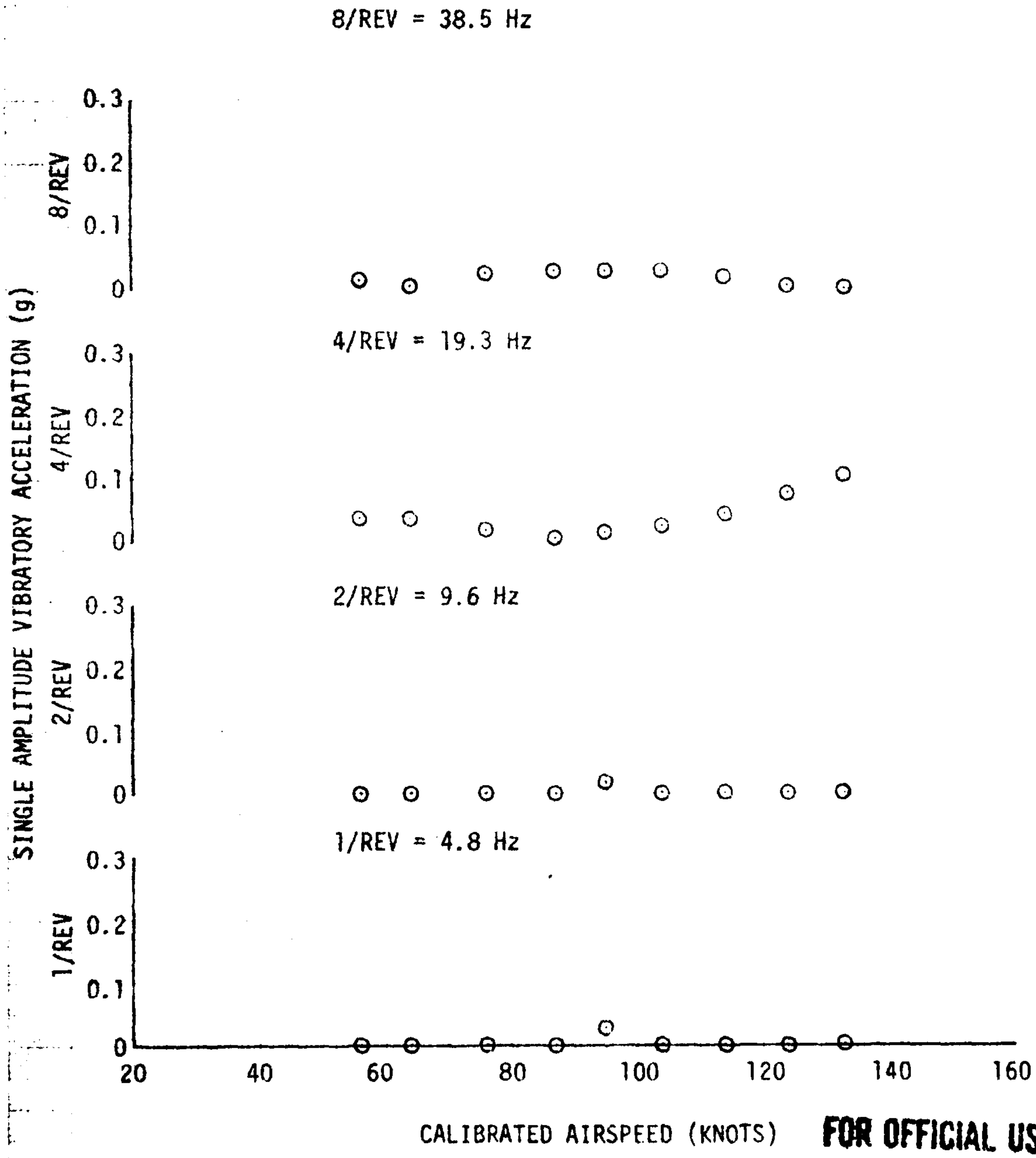

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FIGURE 125
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
AIRCRAFT CG LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

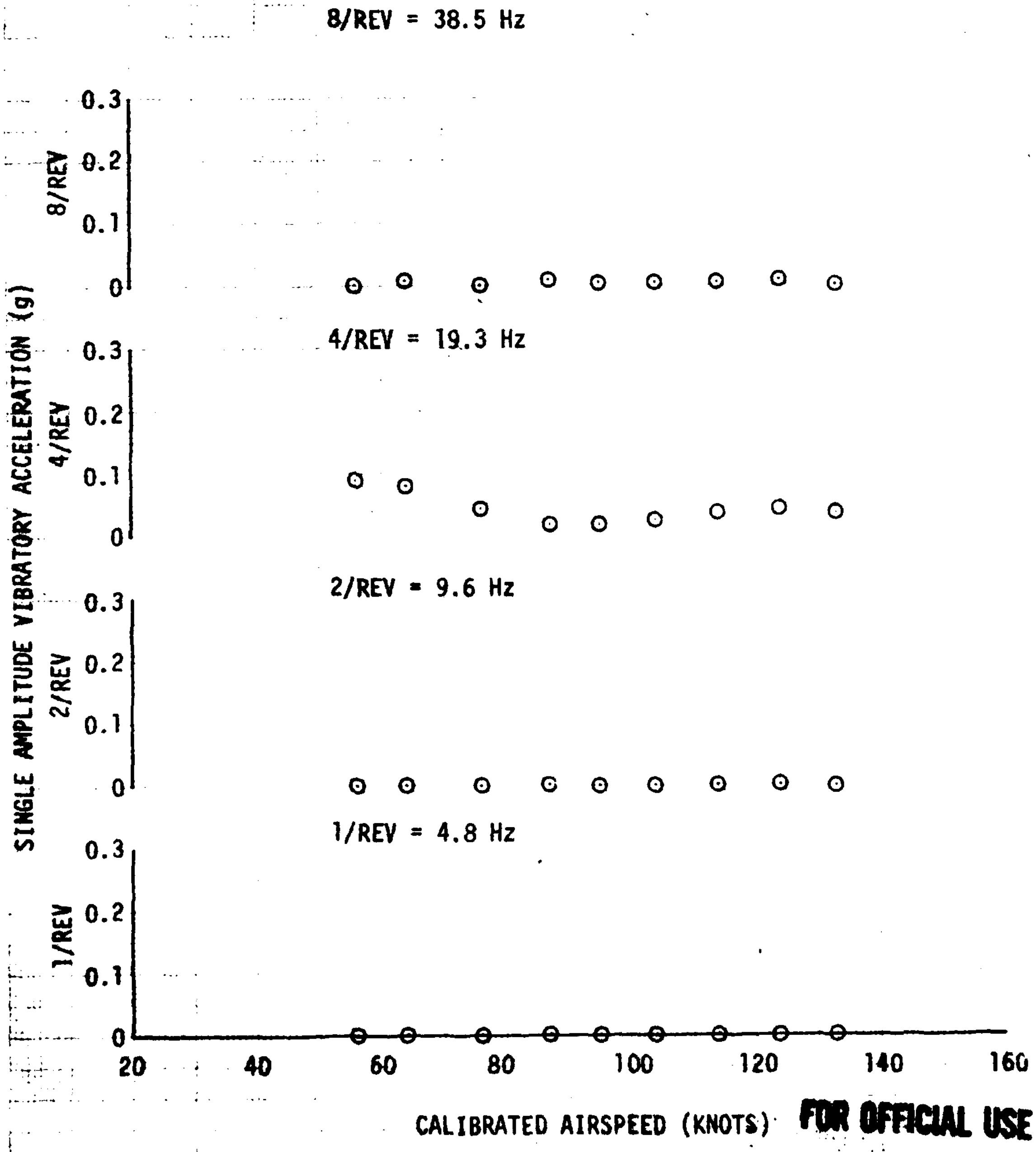

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FIGURE 126
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
AIRCRAFT CG LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

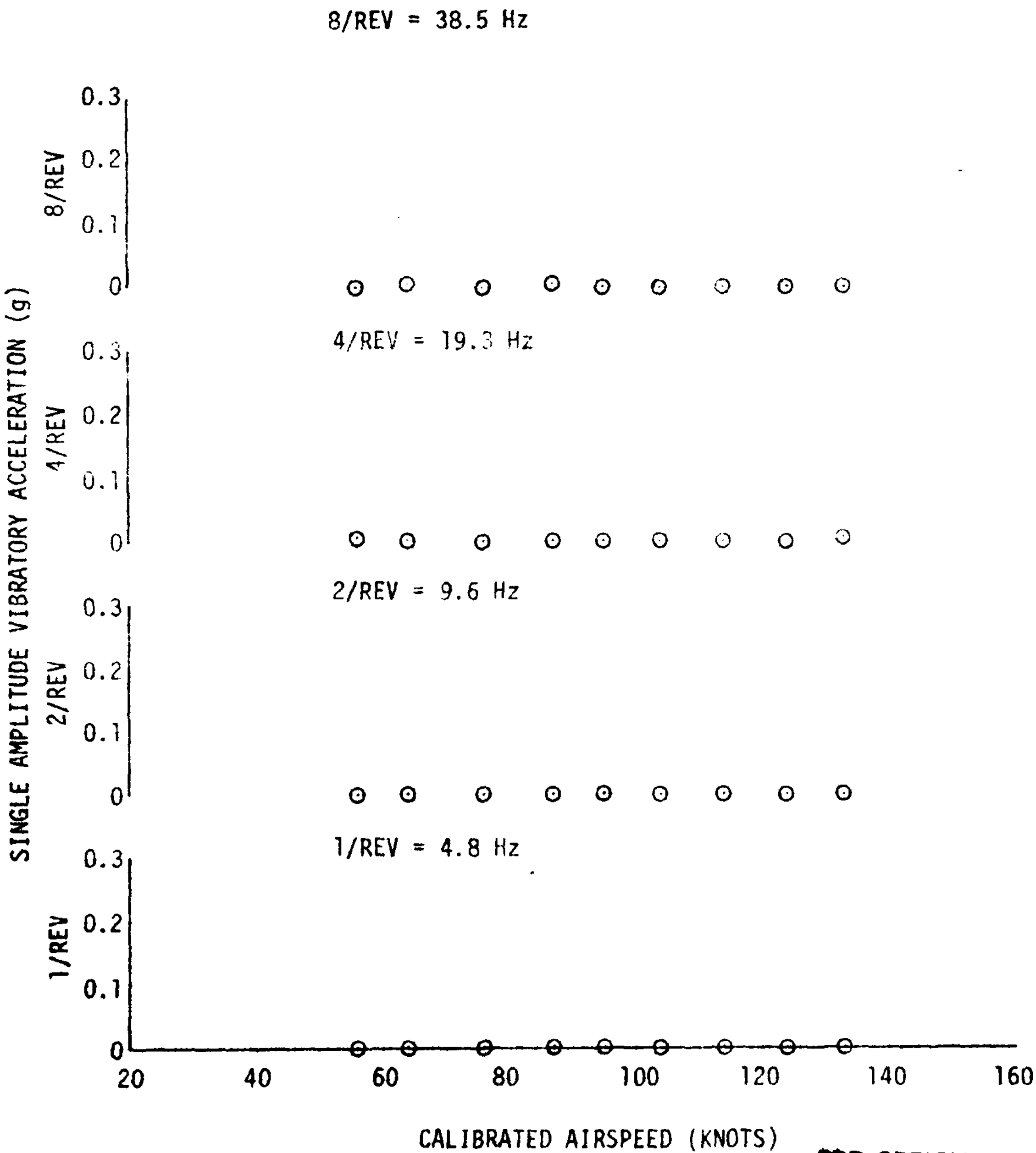

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FIGURE 127
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
LEFT AVIONICS BAY VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

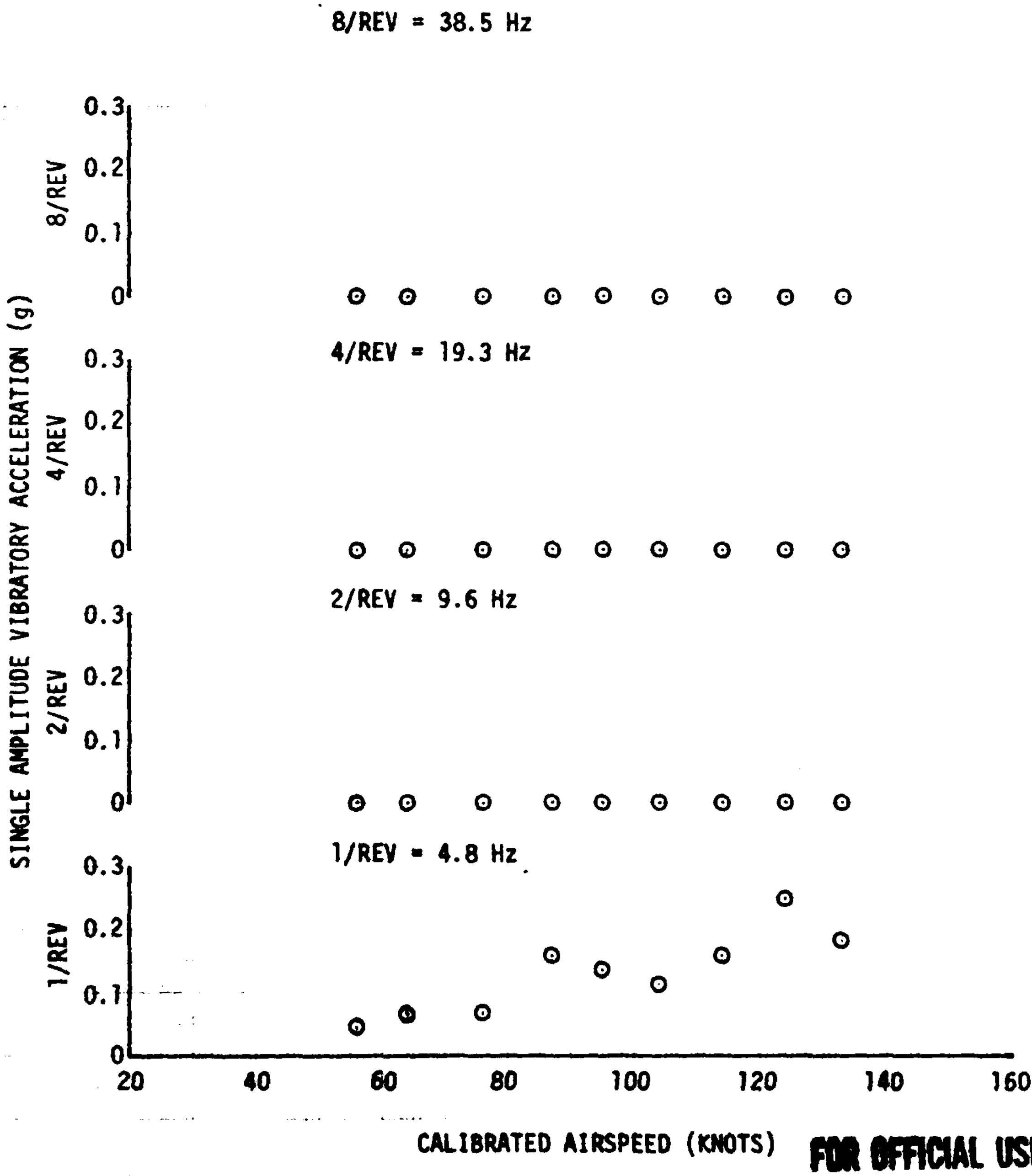

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FIGURE 128
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
LEFT AVIONICS BAY LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

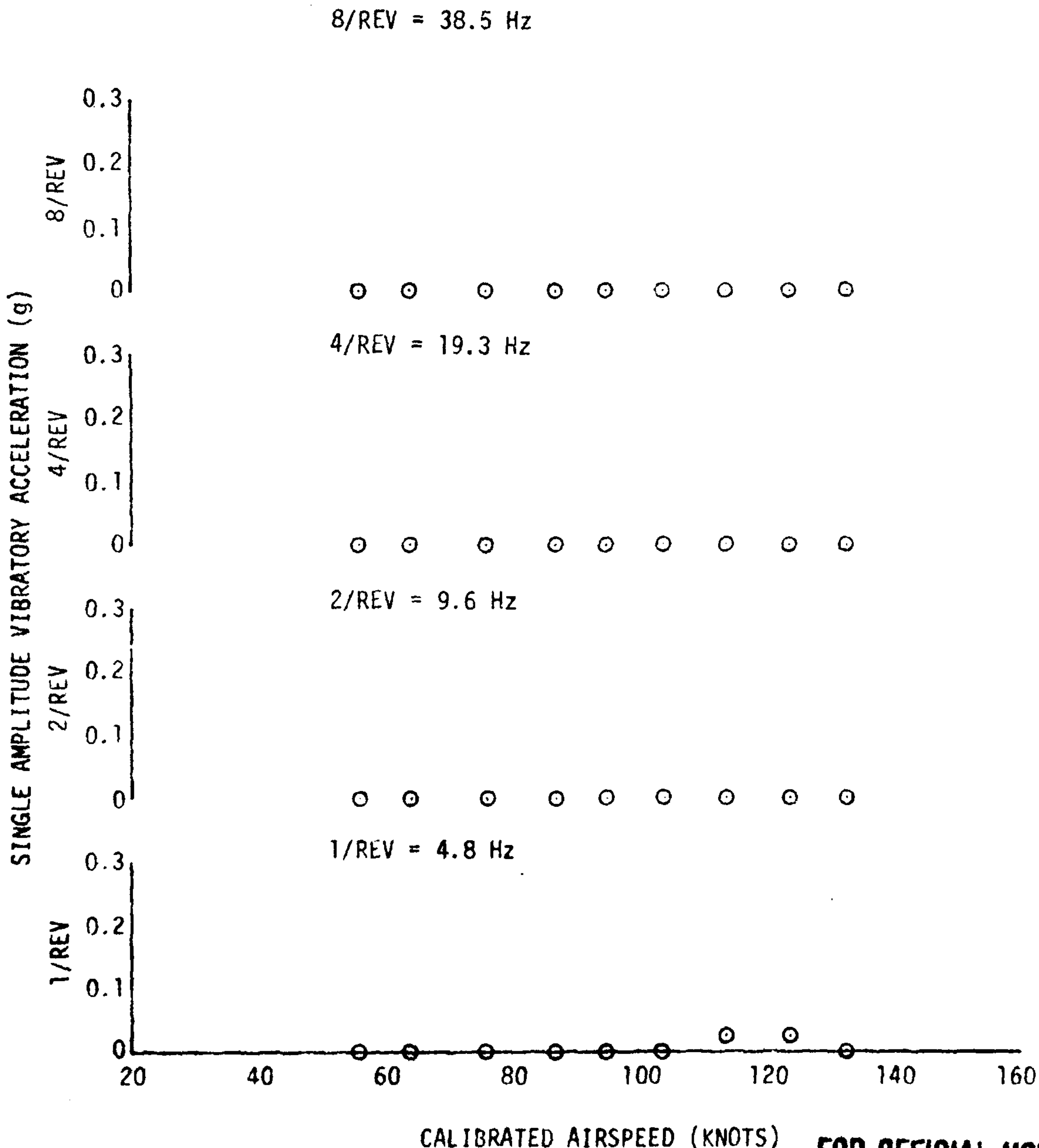

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FIGURE 129
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
LEFT AVIONICS BAY LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

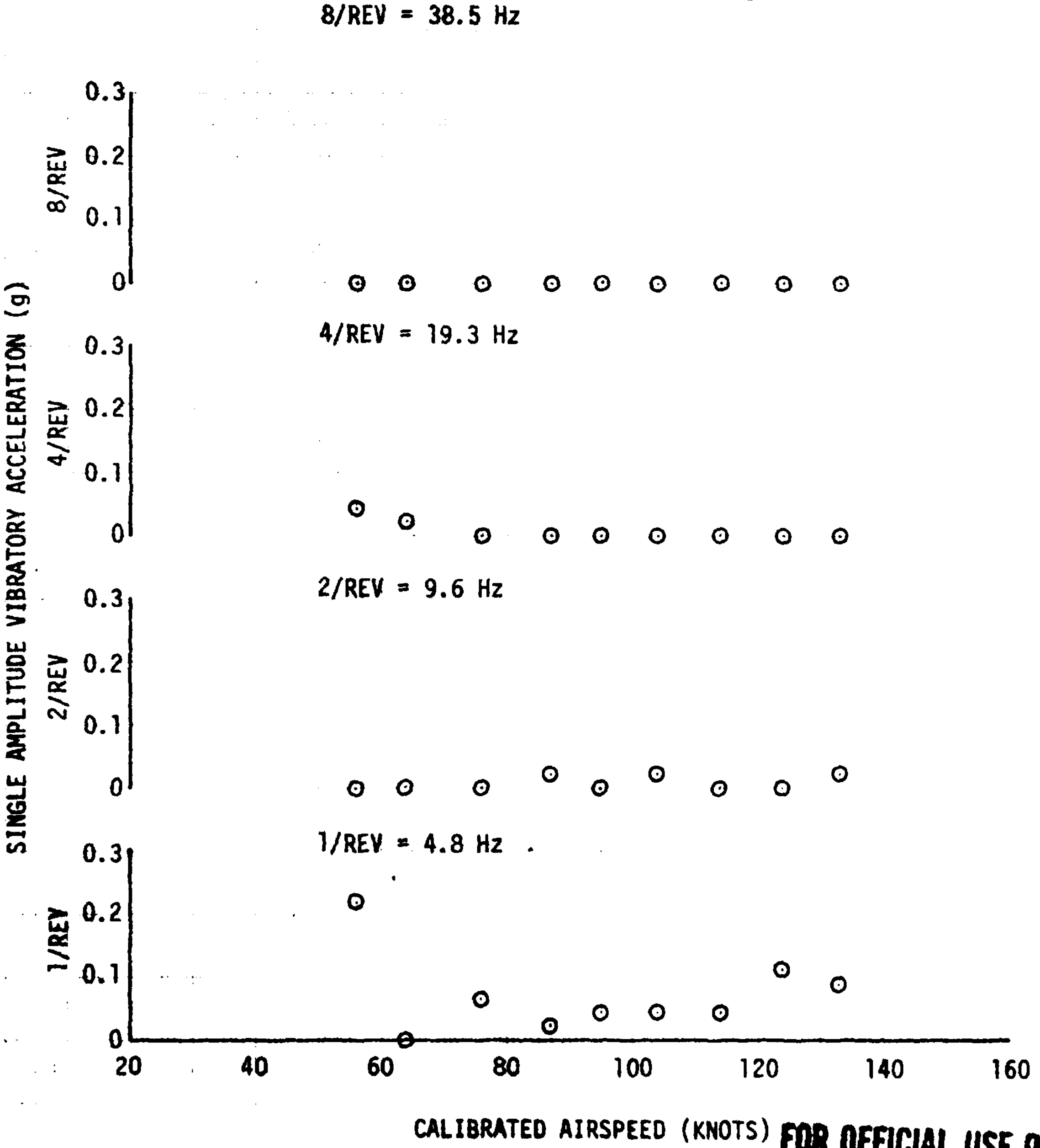

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FIGURE 130
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
RIGHT AVIONICS BAY VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8 TOW

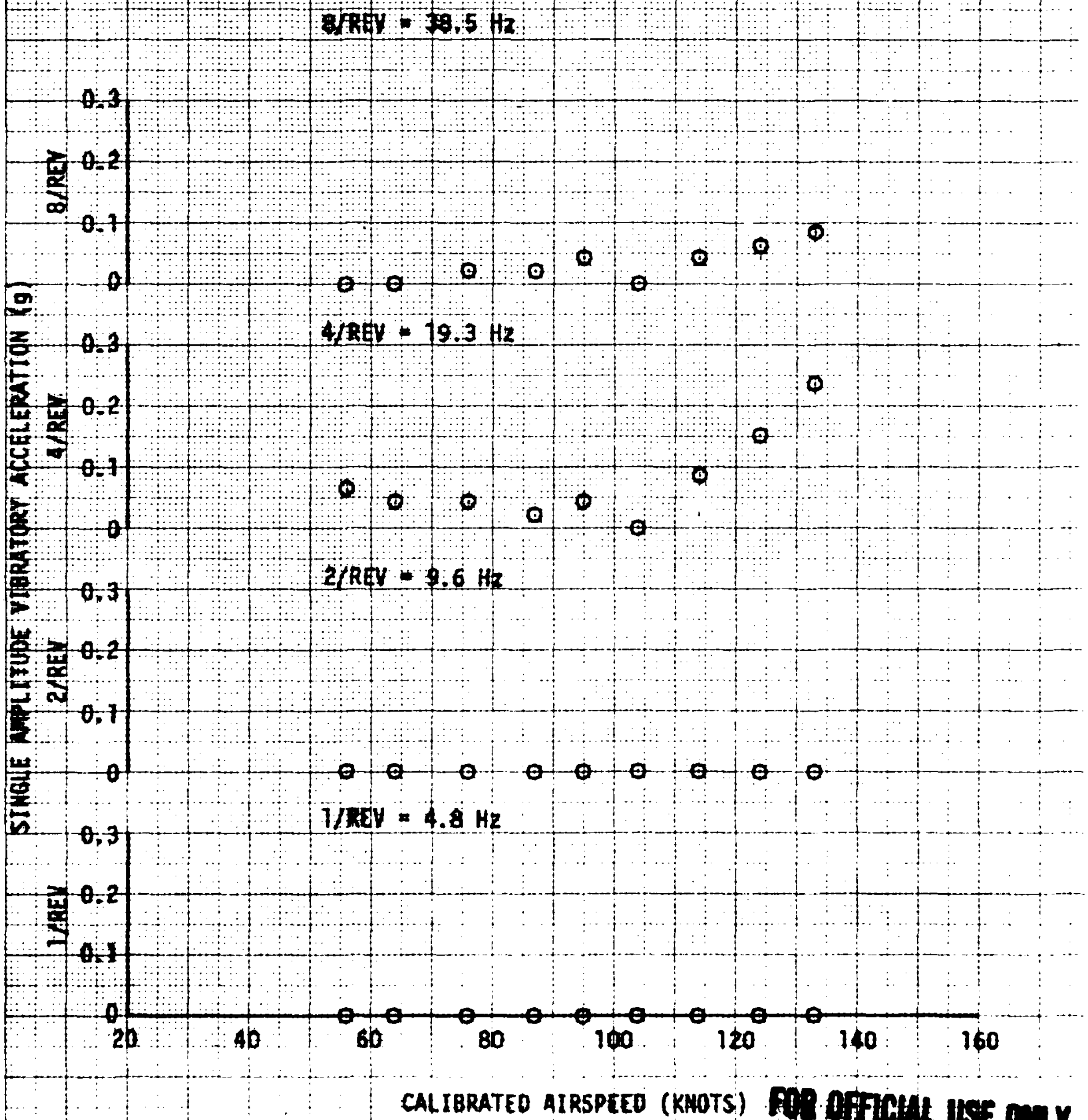

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FIGURE 131
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
RIGHT AVIONICS BAY LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

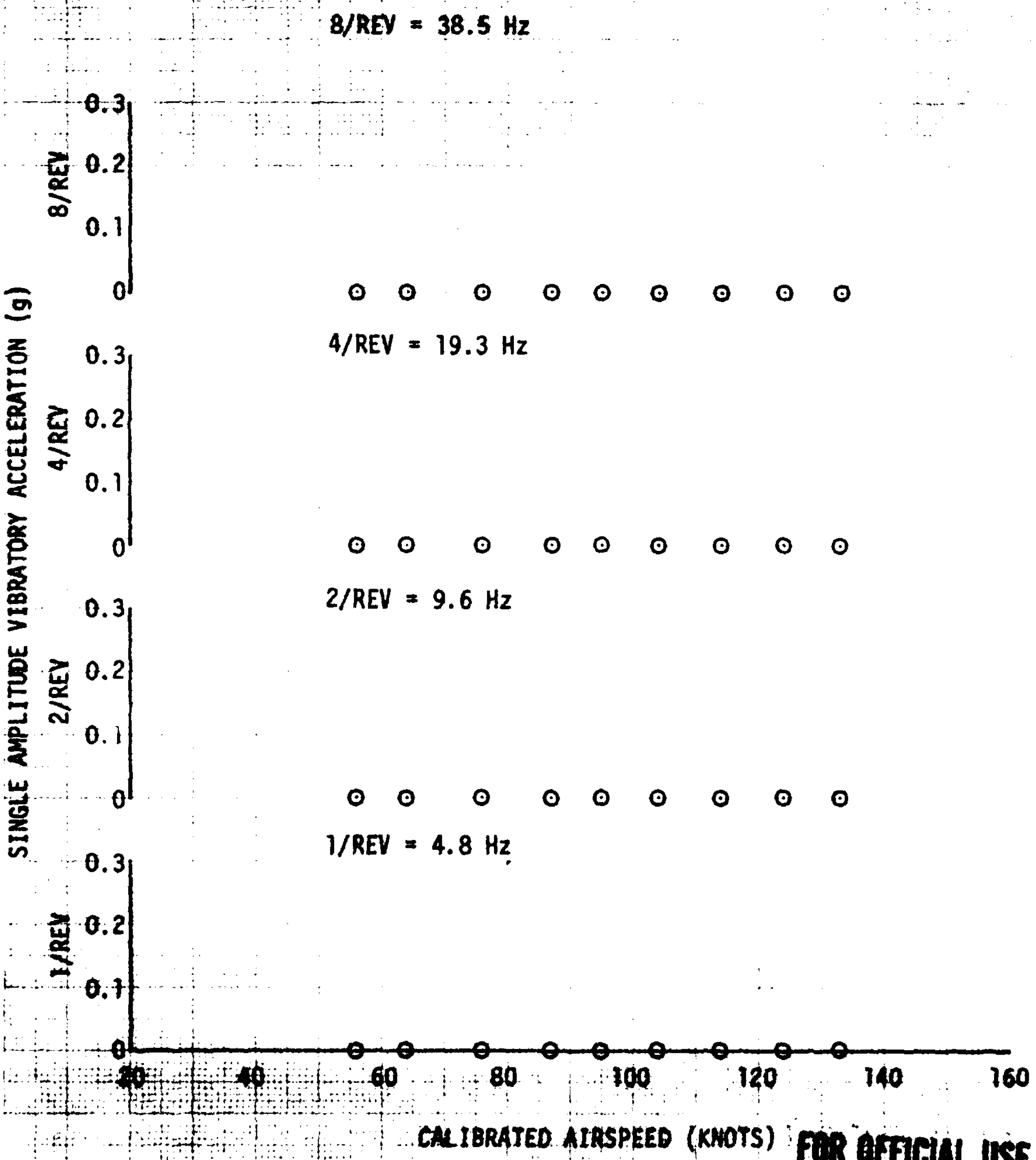

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FIGURE 132
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
RIGHT AVIONICS BAY LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

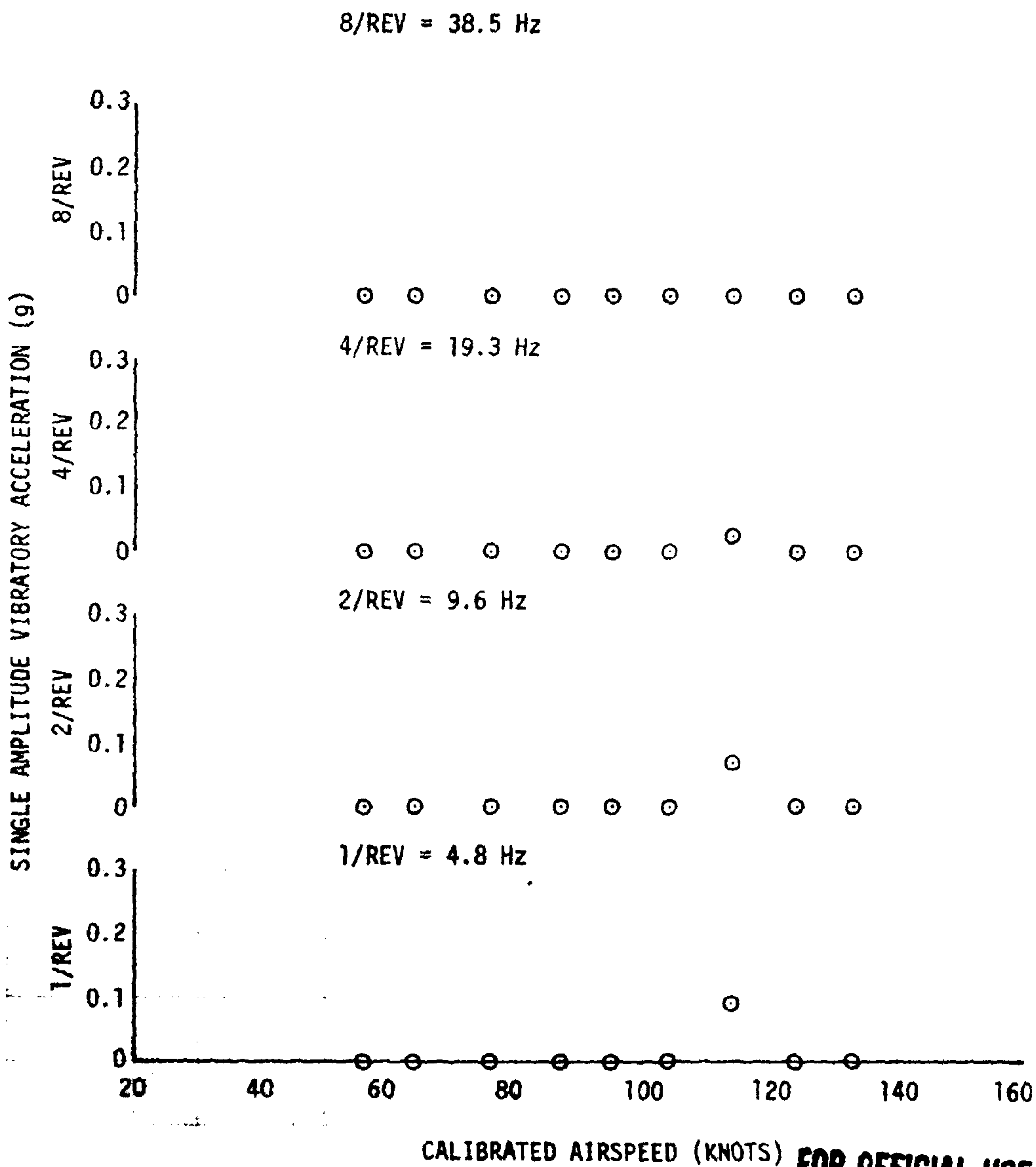

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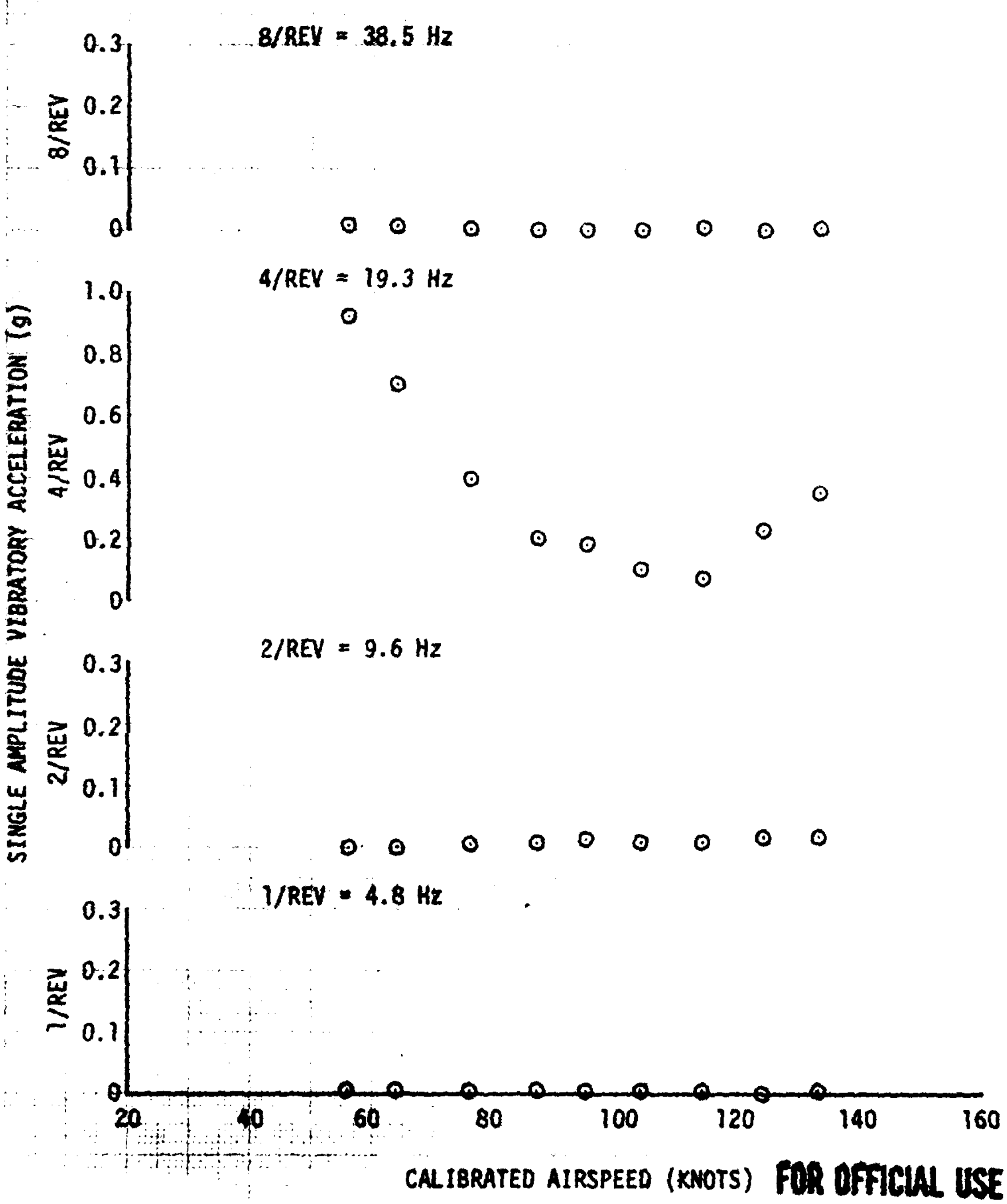
FIGURE 133
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
LEFT WING VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW



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VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
NO. 1 ENGINE EXHAUST VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

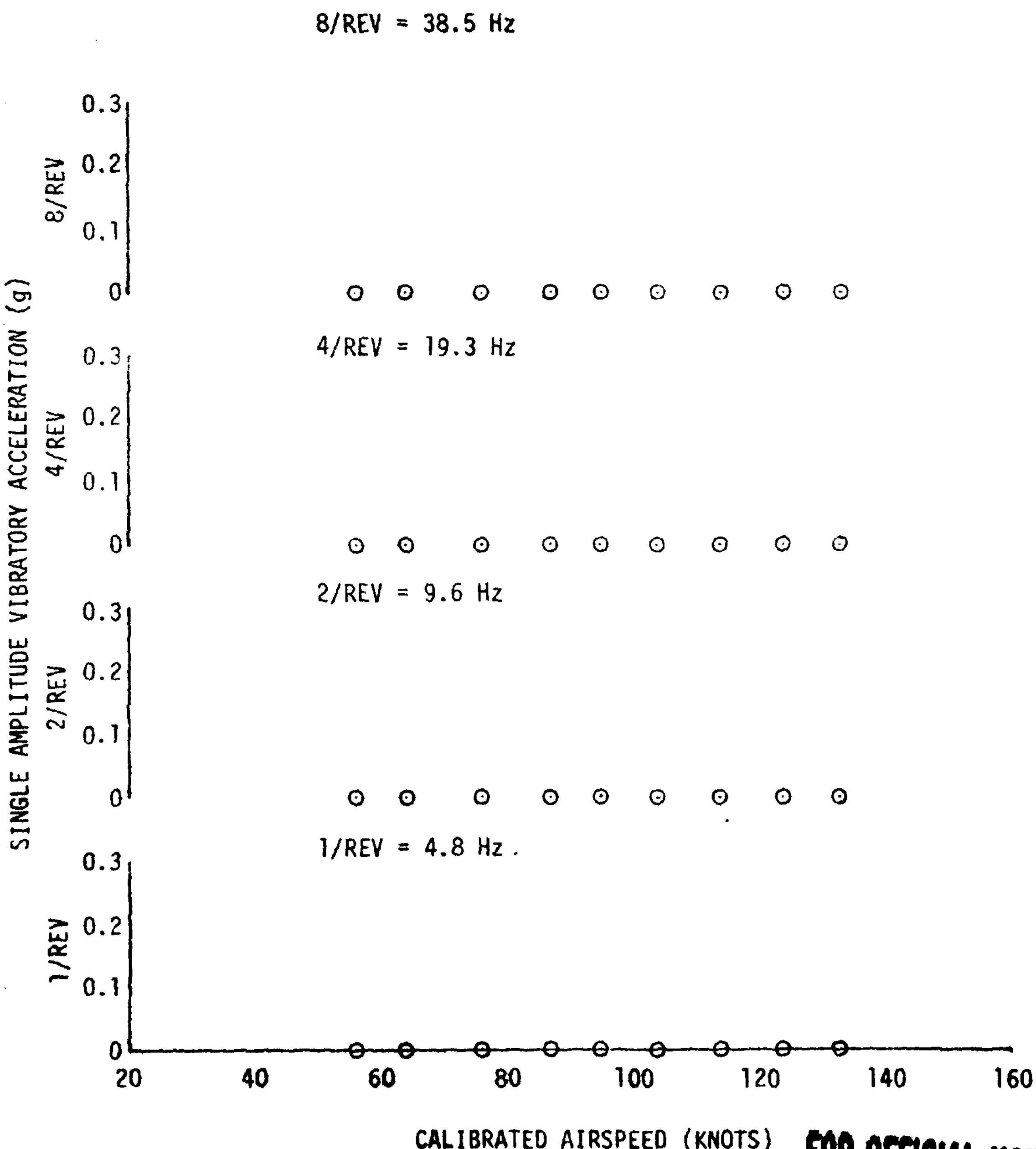

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FIGURE 135
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
MAIN TRANSMISSION VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

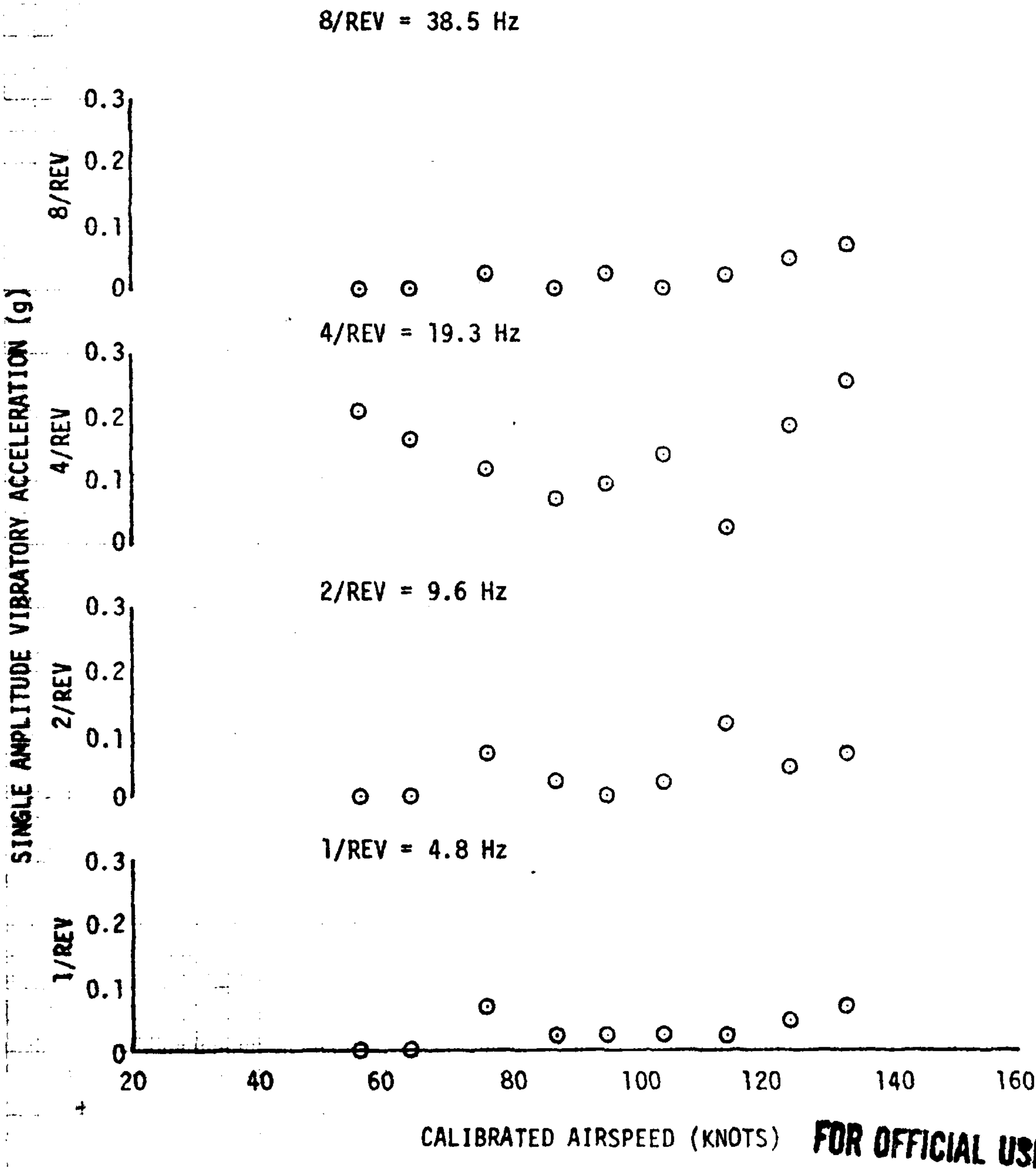

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FIGURE 130
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
TAIL ROTOR GEAR BOX VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
13650	204.4(MID)	0.00(MID)	5000	25.0	287	LEVEL FLT	8-TOW

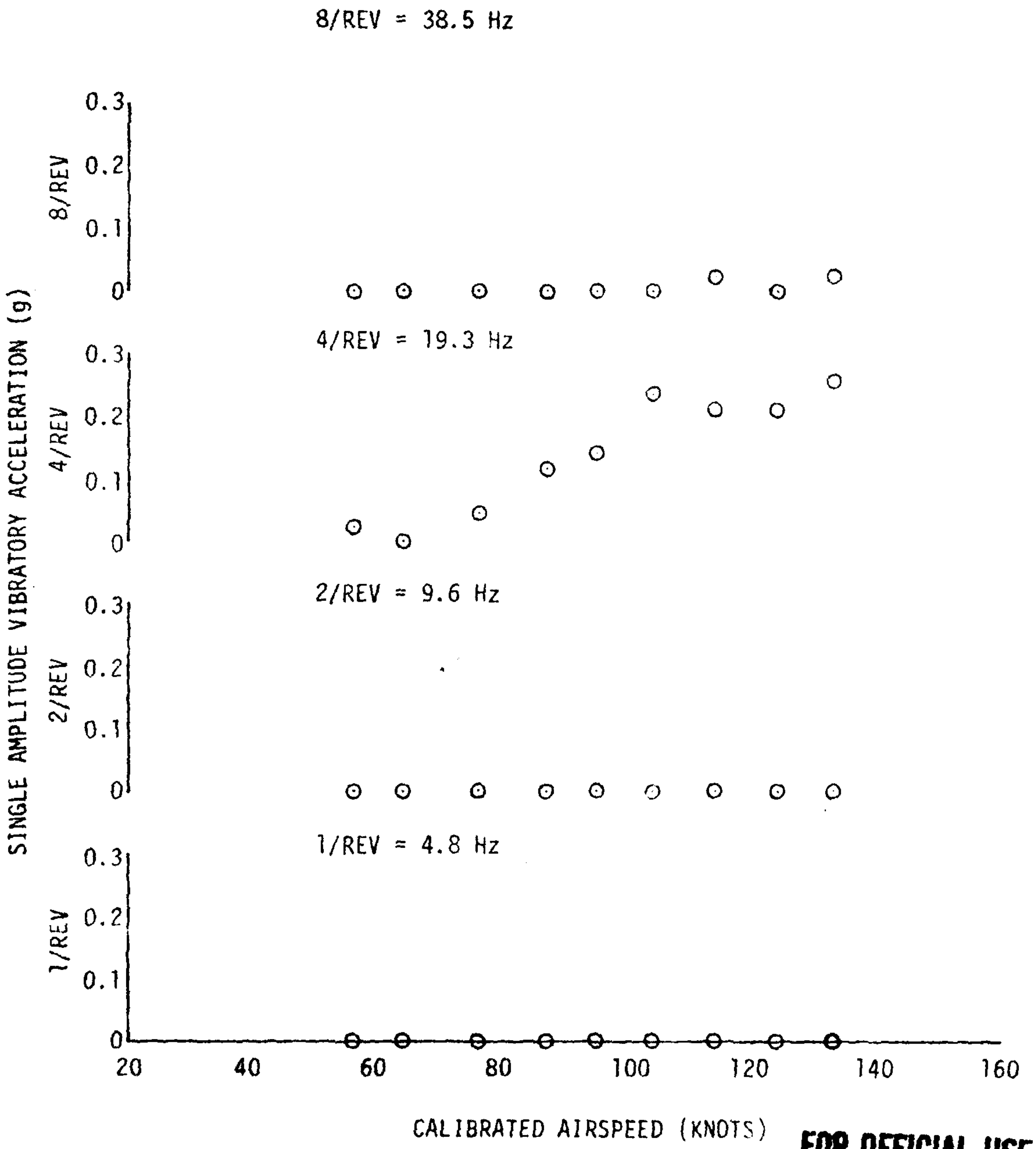

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FIGURE 137
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
PILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
15800	200.0(FWD)	0 (MID)	7400	16.0	290	LVL FLT	8 TOW

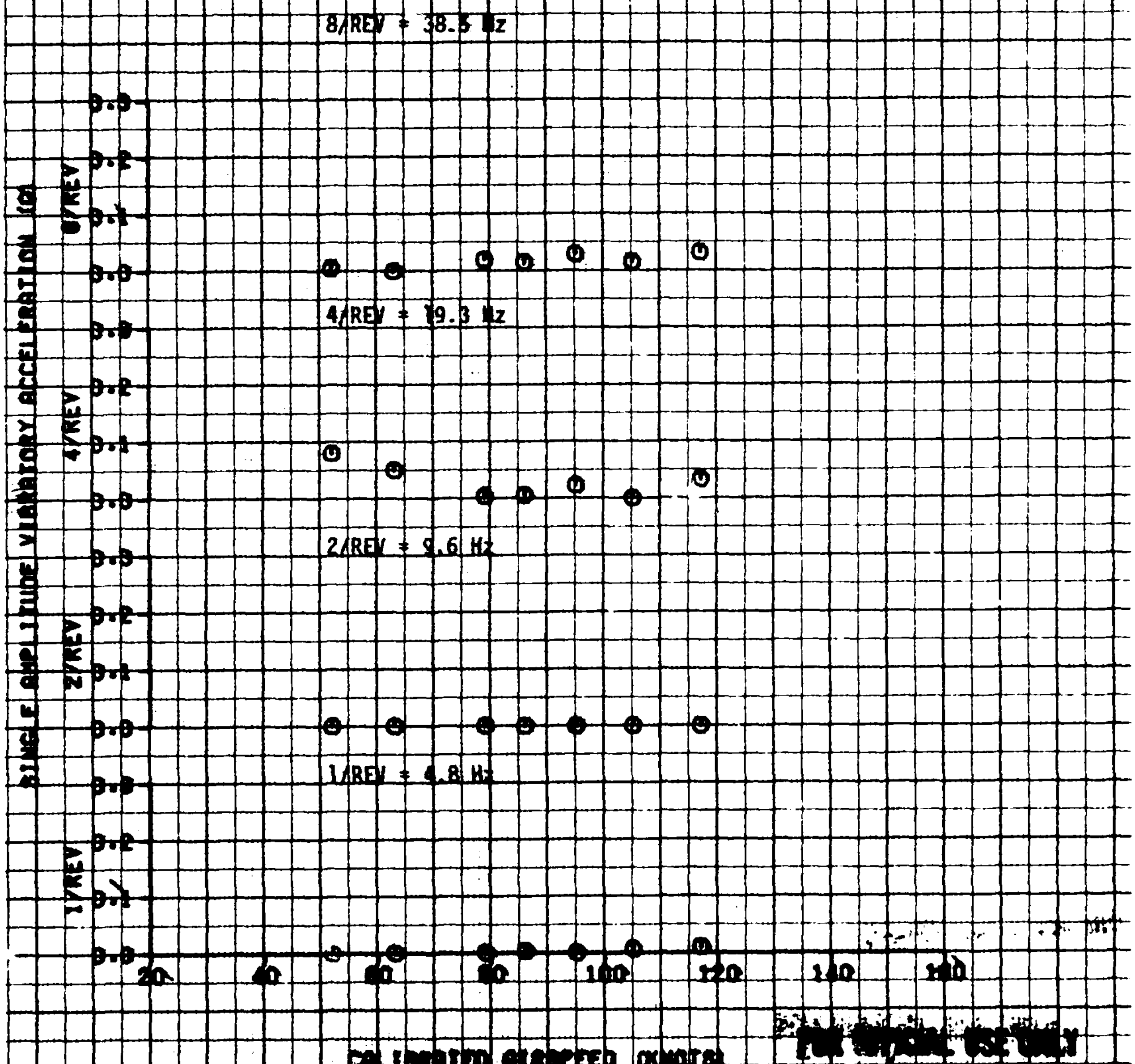


FIGURE 138
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
PILOT SEAT LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
15800	200.0(FWD)	0 (MID)	7400	18.0	290	LVL FLT	8 TOW

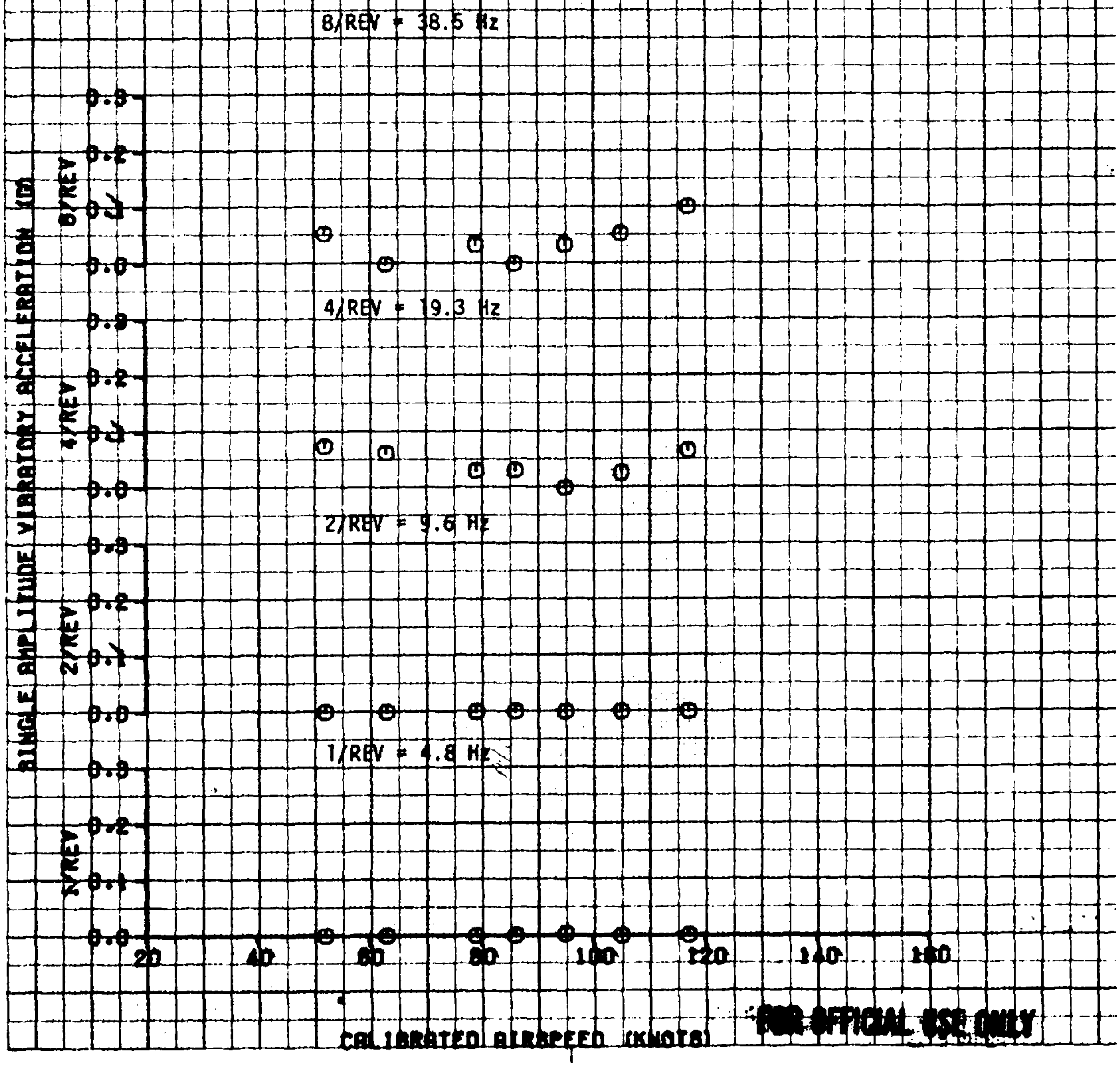


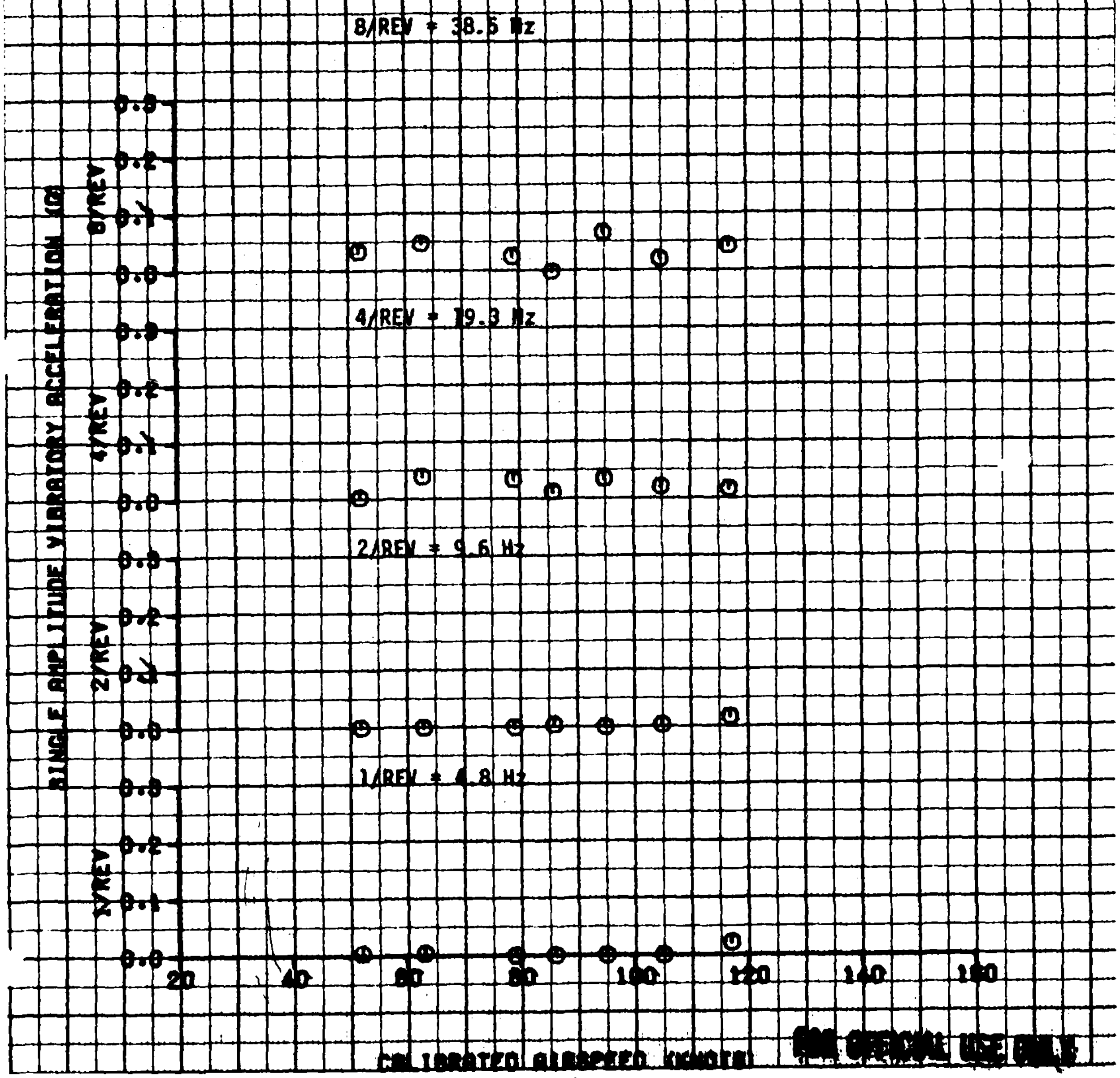
FIGURE 139

VIBRATION CHARACTERISTICS

YAH-64 USA S/N 74-22249

COPILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
15500	200.0 (FWD)	0 (MID)	7400	18.0	289	LVL FLT	[illegible]



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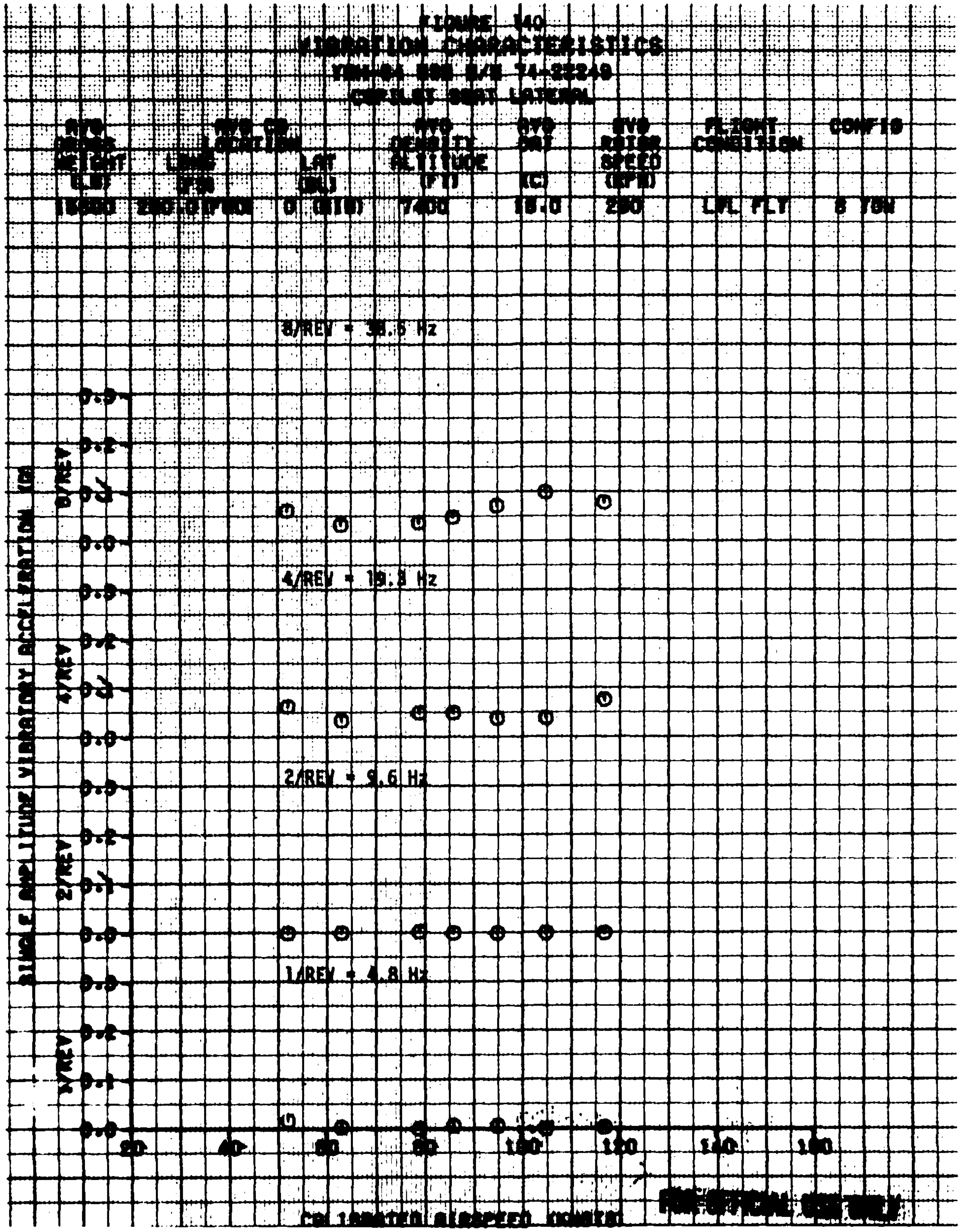


FIGURE 140
VIBRATION CHARACTERISTICS
LATERAL
FLIGHT CONDITION
CONFIG
ALTITUDE (FT)
SPEED (RPM)
(C)
LAT
7400
250
LVL FLT
8/REV = 38.5 Hz
4/REV = 19.3 Hz
2/REV = 9.6 Hz
1/REV = 4.8 Hz
8/REV
4/REV
2/REV
1/REV
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

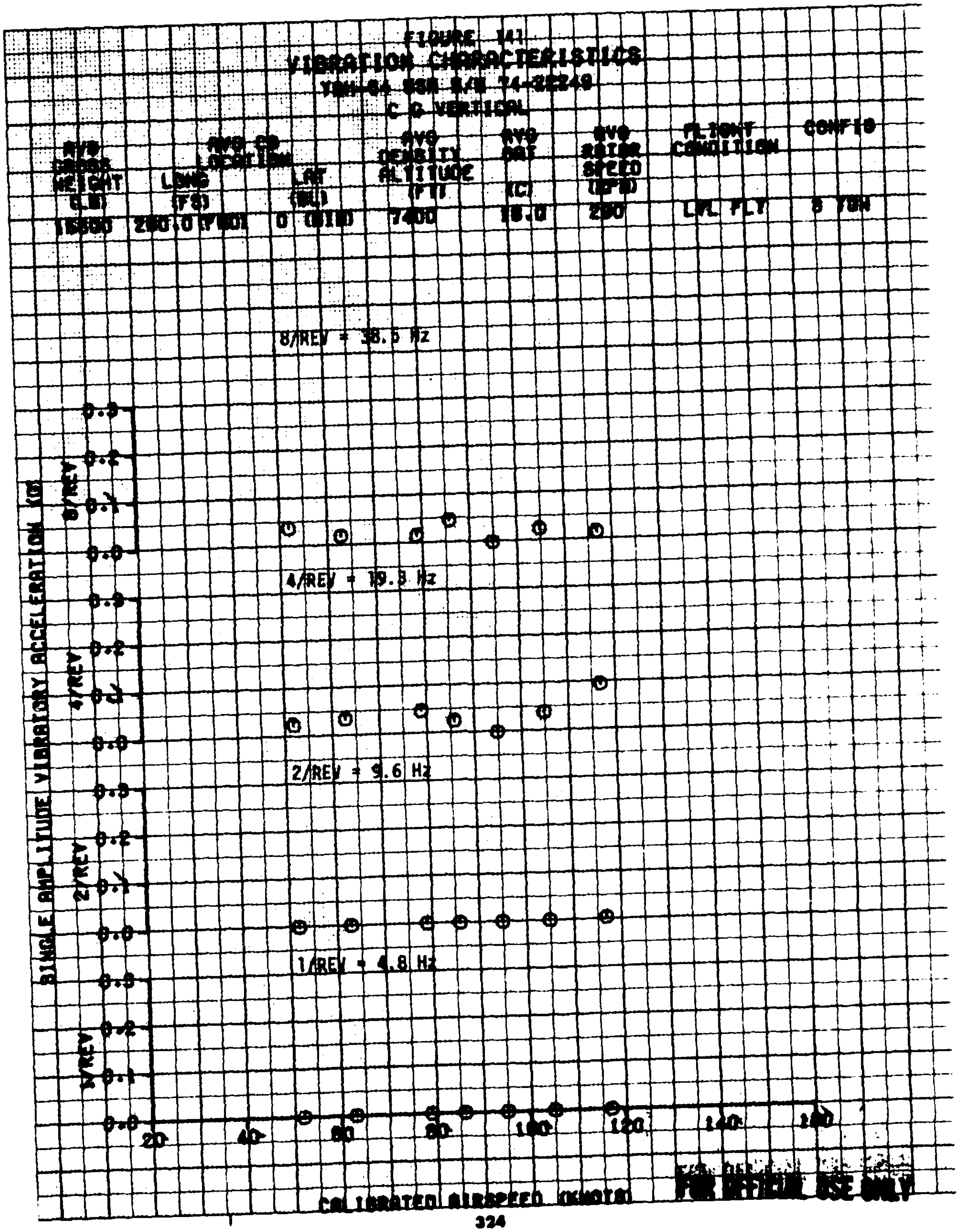
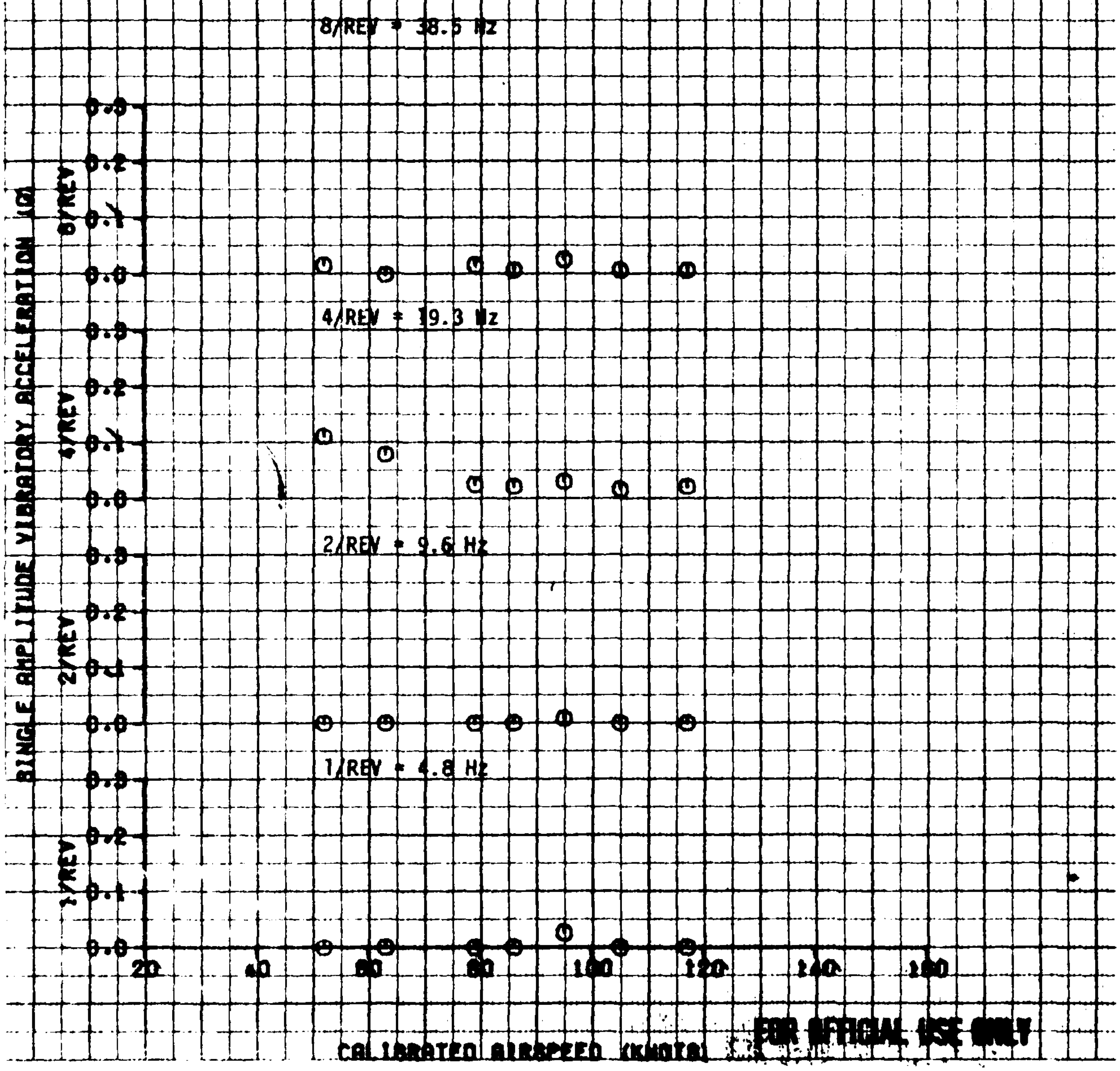


FIGURE 141
VIBRATION CHARACTERISTICS
C G VERTICAL
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
CONFIG
0 (MID)
7400
290
LVL FLY
8/REV = 38.5 Hz
4/REV = 19.3 Hz
2/REV = 9.6 Hz
1/REV = 4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
8/REV
4/REV
2/REV
1/REV
0.3
0.2
0.1
0.0
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

FIGURE 142
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
C G LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
15800	200.0 (FWD)	0 (MID)	7400	18.0	290	LVL FLT	8 TOW



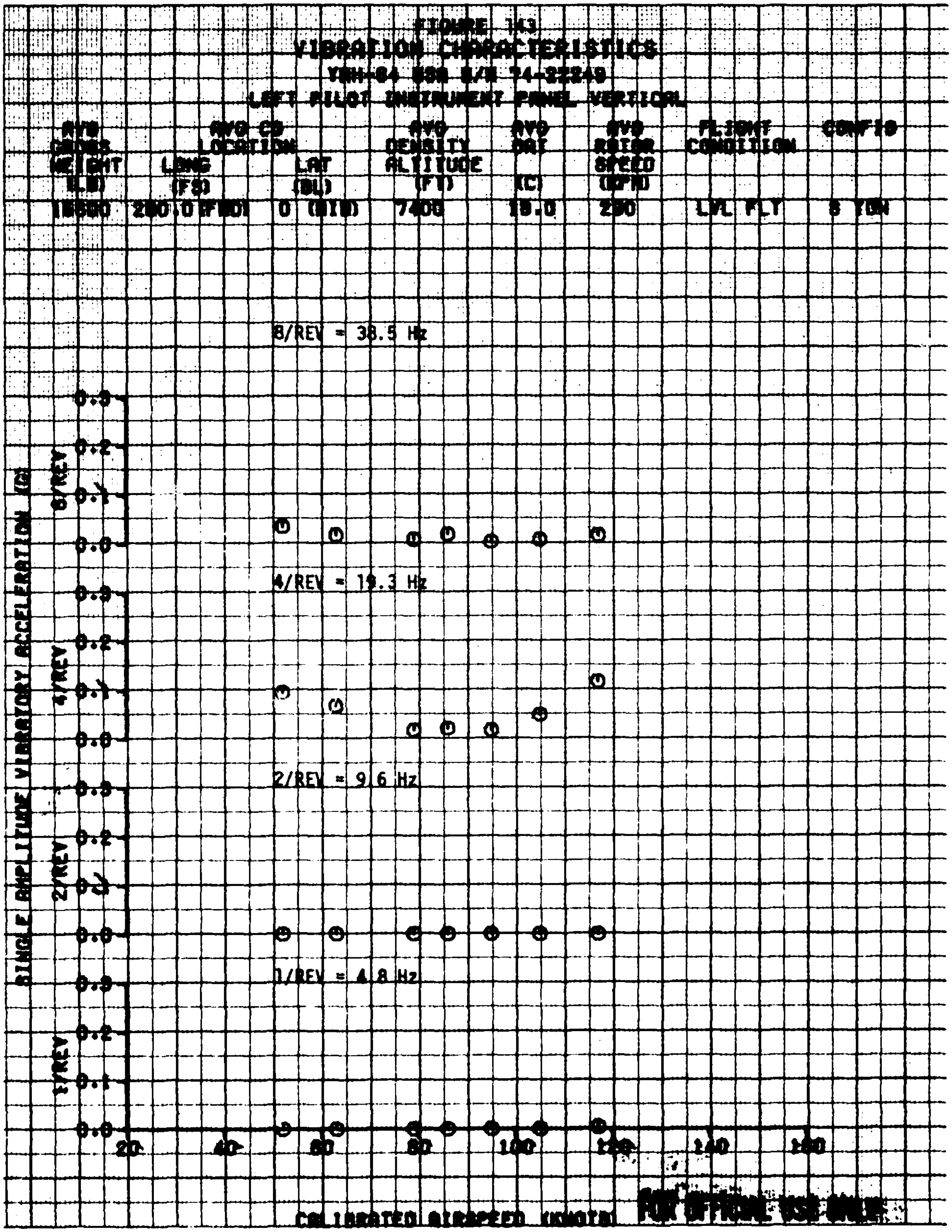


FIGURE 143
VIBRATION CHARACTERISTICS
LEFT PILOT INSTRUMENT PANEL VERTICAL
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
CONFIG
7400
290
LVL FLT
8/REV = 38.5 Hz
4/REV = 19.3 Hz
2/REV = 9.6 Hz
1/REV = 4.8 Hz
8/REV
4/REV
2/REV
1/REV
0.0
0.1
0.2
0.3
20
40
60
80
100
120
140
160
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
CALIBRATED AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

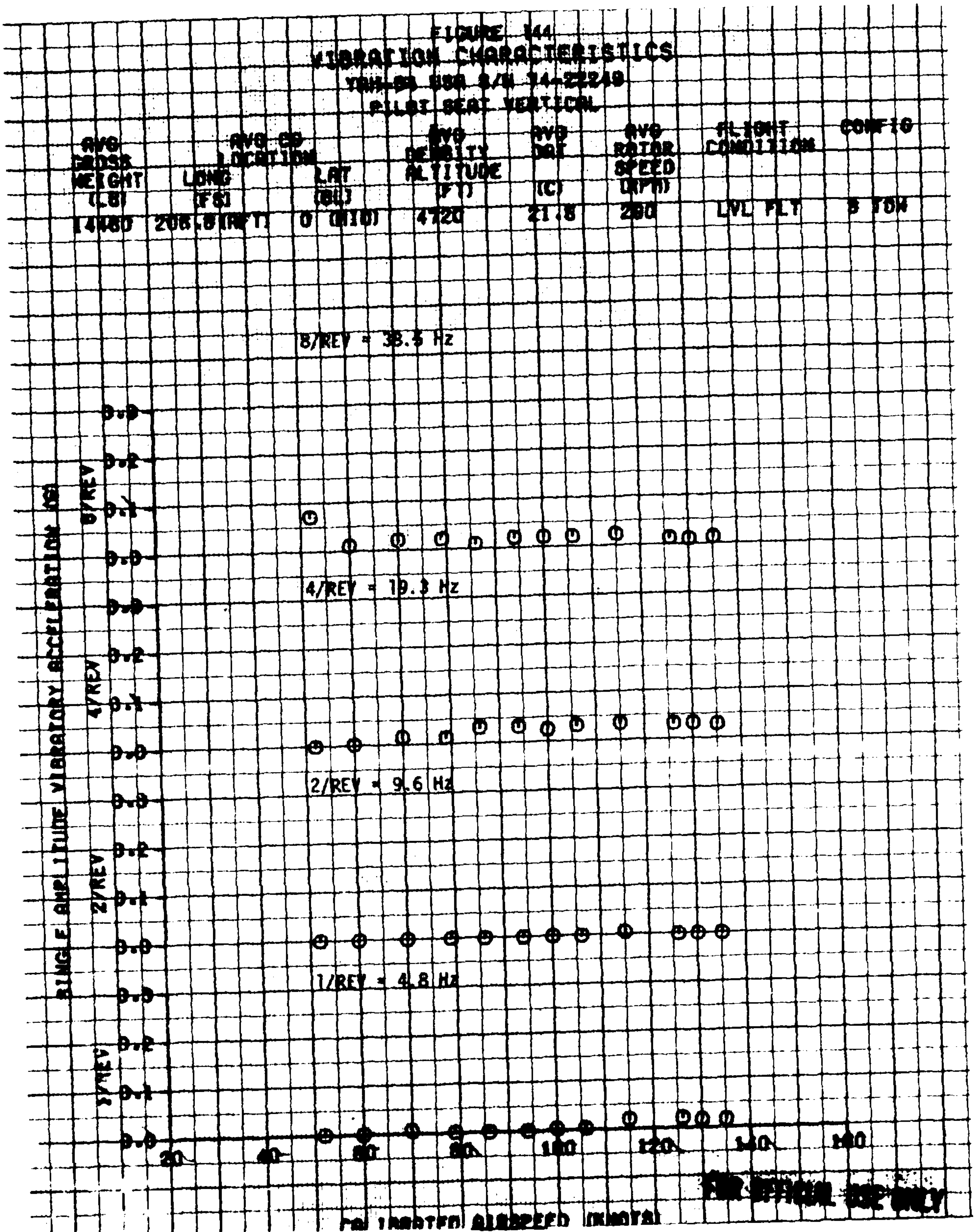


FIGURE 144
VIBRATION CHARACTERISTICS
YUH-60 USA S/N 74-22249
PILOT SEAT VERTICAL
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
CONFIG
14460
208.8(AFT)
0 (MID)
4720
21.8
290
LVL FLT
B TOW
8/REV = 38.5 Hz
4/REV = 19.3 Hz
2/REV = 9.6 Hz
1/REV = 4.8 Hz
8/REV
4/REV
2/REV
1/REV
0.3
0.2
0.1
0.0
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

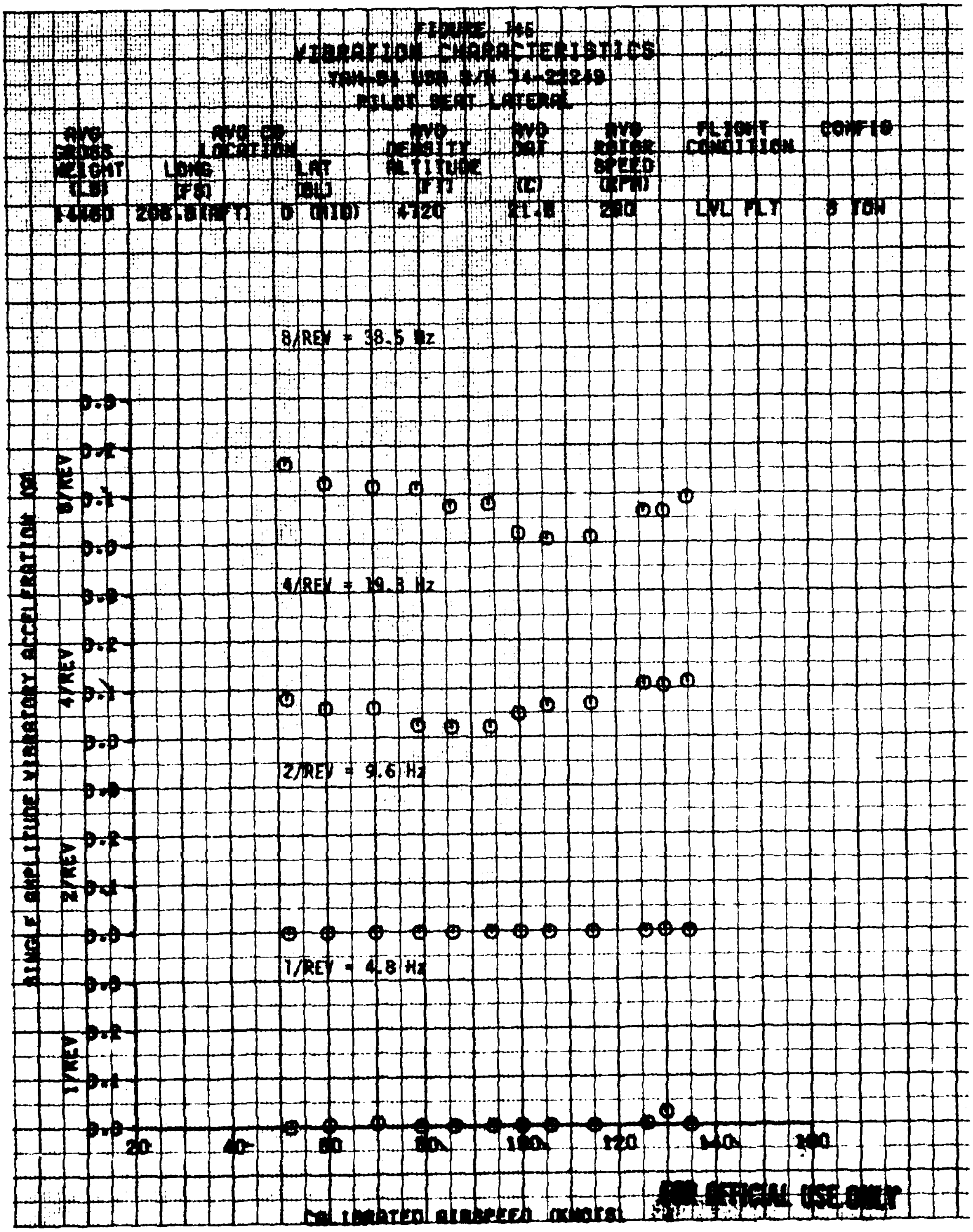
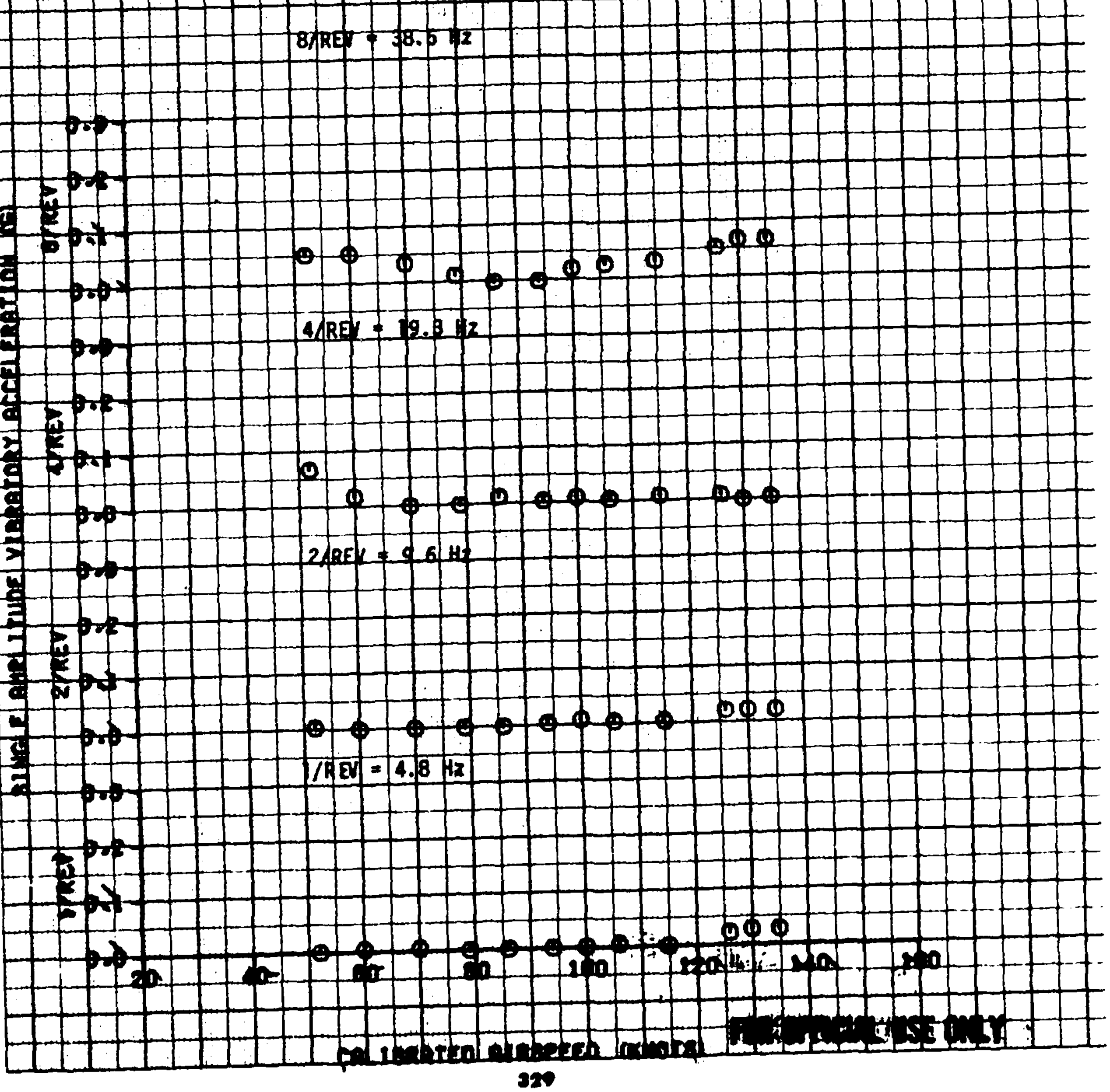

VIBRATION CHARACTERISTICS
PILOT SEAT LATERAL
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
CONFIG
LVL FLT
8/REV = 38.5 Hz
4/REV = 19.3 Hz
2/REV = 9.6 Hz
1/REV = 4.8 Hz
8/REV
4/REV
2/REV
1/REV
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)
FOR OFFICIAL USE ONLY

FIGURE 146
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
COPILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
14460	206.8(AFT)	0 (MID)	4720	21.5	289	LVL FLT	8 TOW



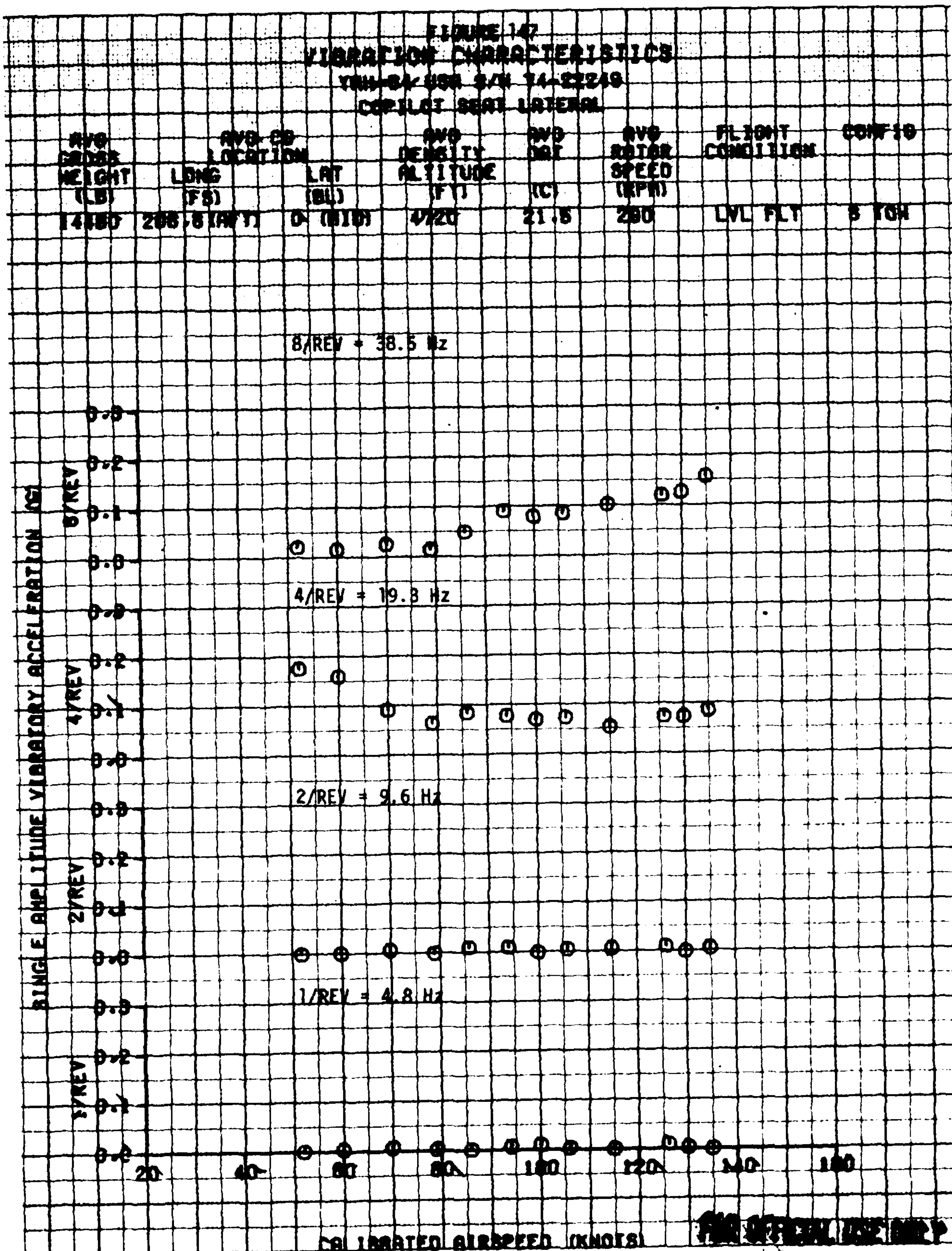


FIGURE 147
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
COPILOT SEAT LATERAL
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
CONFIG
14480
206.8 (AFT)
0 (MID)
4720
21.5
290
LVL FLT
8 TOW
8/REV = 38.5 Hz
4/REV = 19.3 Hz
2/REV = 9.6 Hz
1/REV = 4.8 Hz
8/REV
4/REV
2/REV
1/REV
0.3
0.2
0.1
0.0
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)

FIGURE 148

VIBRATION CHARACTERISTICS

YAH-64 USA S/N 74-22249

C G VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
14460	206.8(AFT)	0 (MID)	4720	21.5	290	LVL FLT	8 TOW

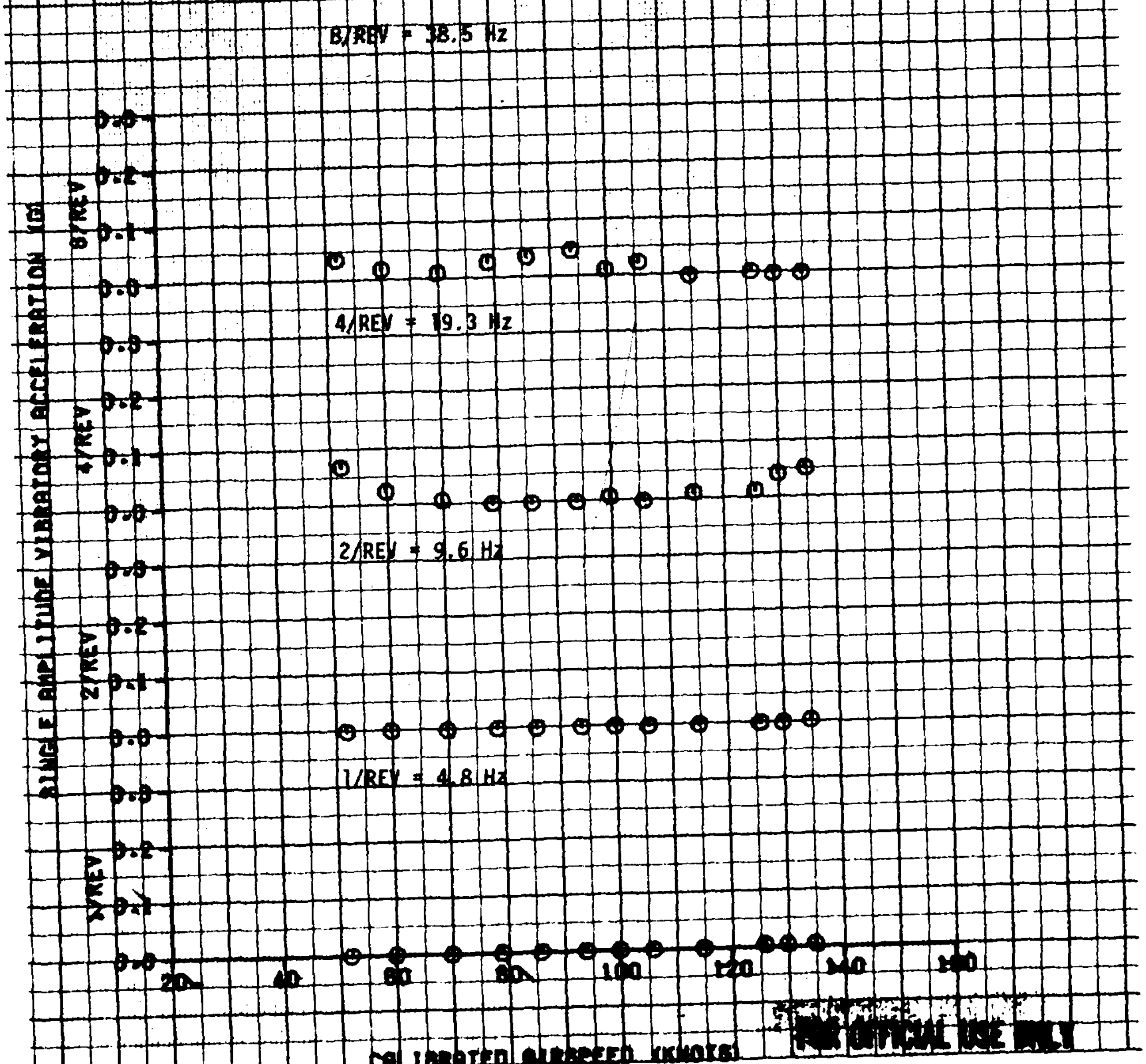


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FIGURE 149
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
C G LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
14480	205.8(AFT)	0 (MID)	4720	21.5	290	LVL FLT	8 TOW

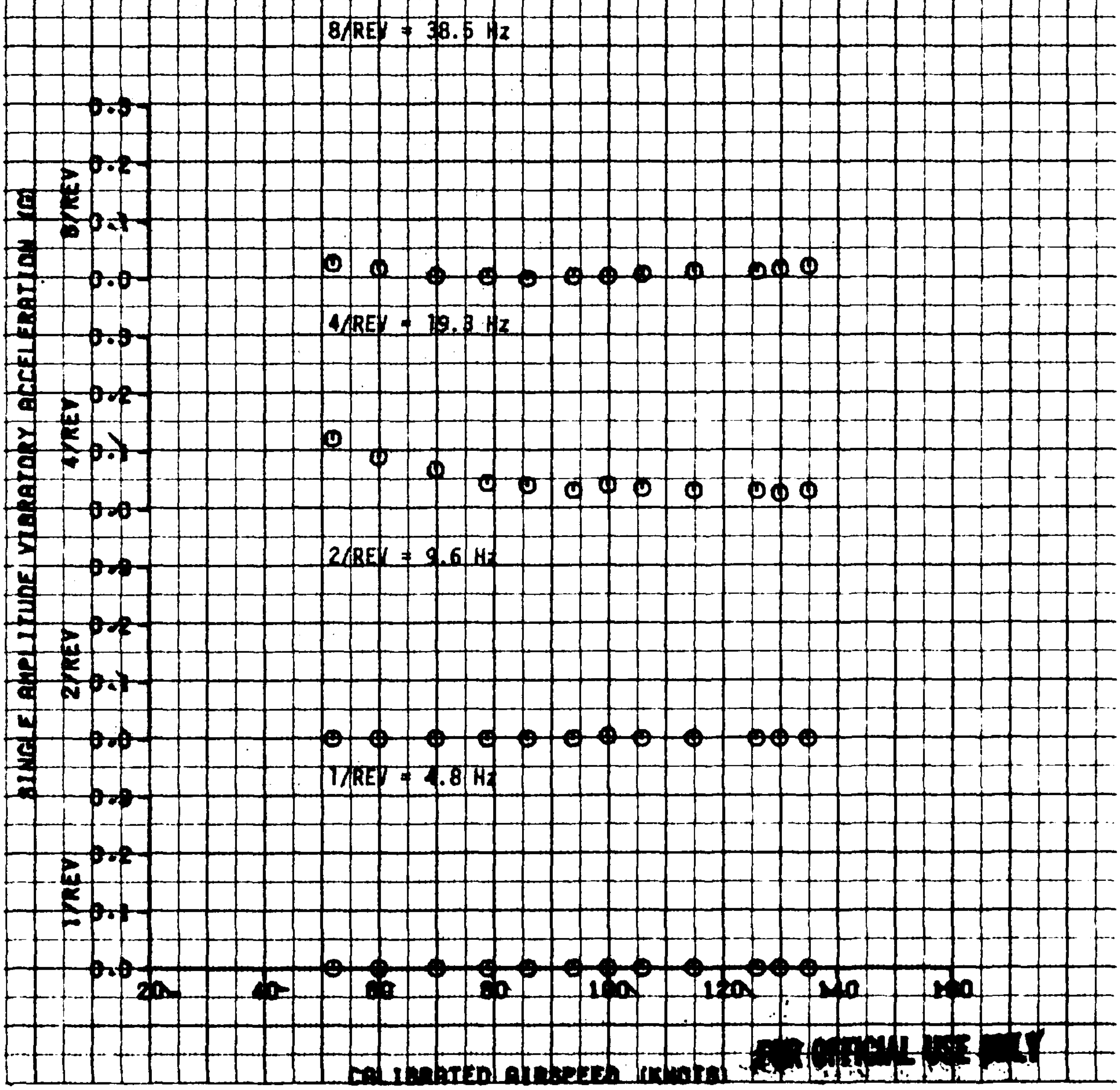

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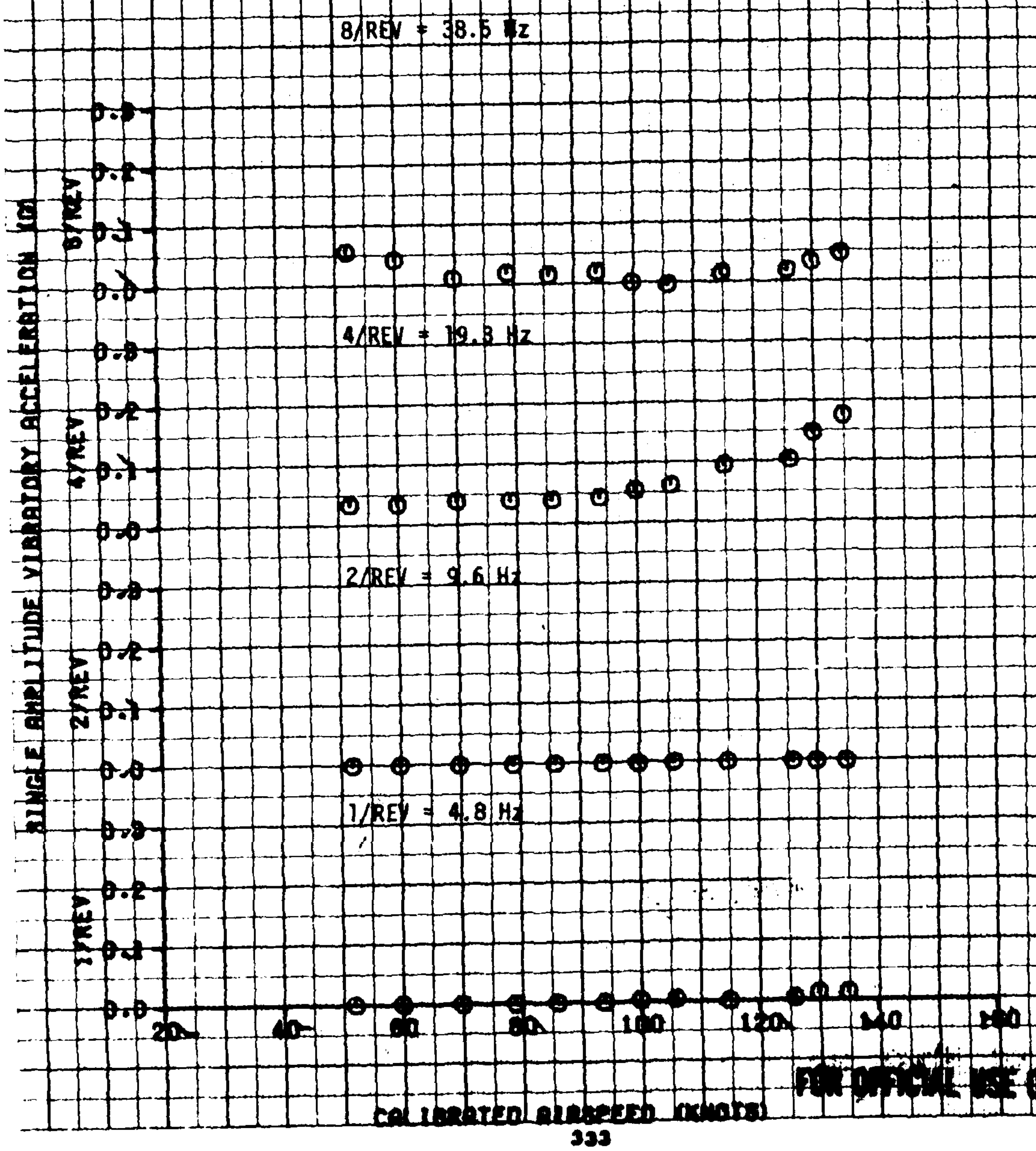
FIGURE 160
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
LEFT PILOT INSTRUMENT PANEL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
14460	206.8 (AFT)	0 (MID)	4720	21.5	290	LVL FLT	8 TOW



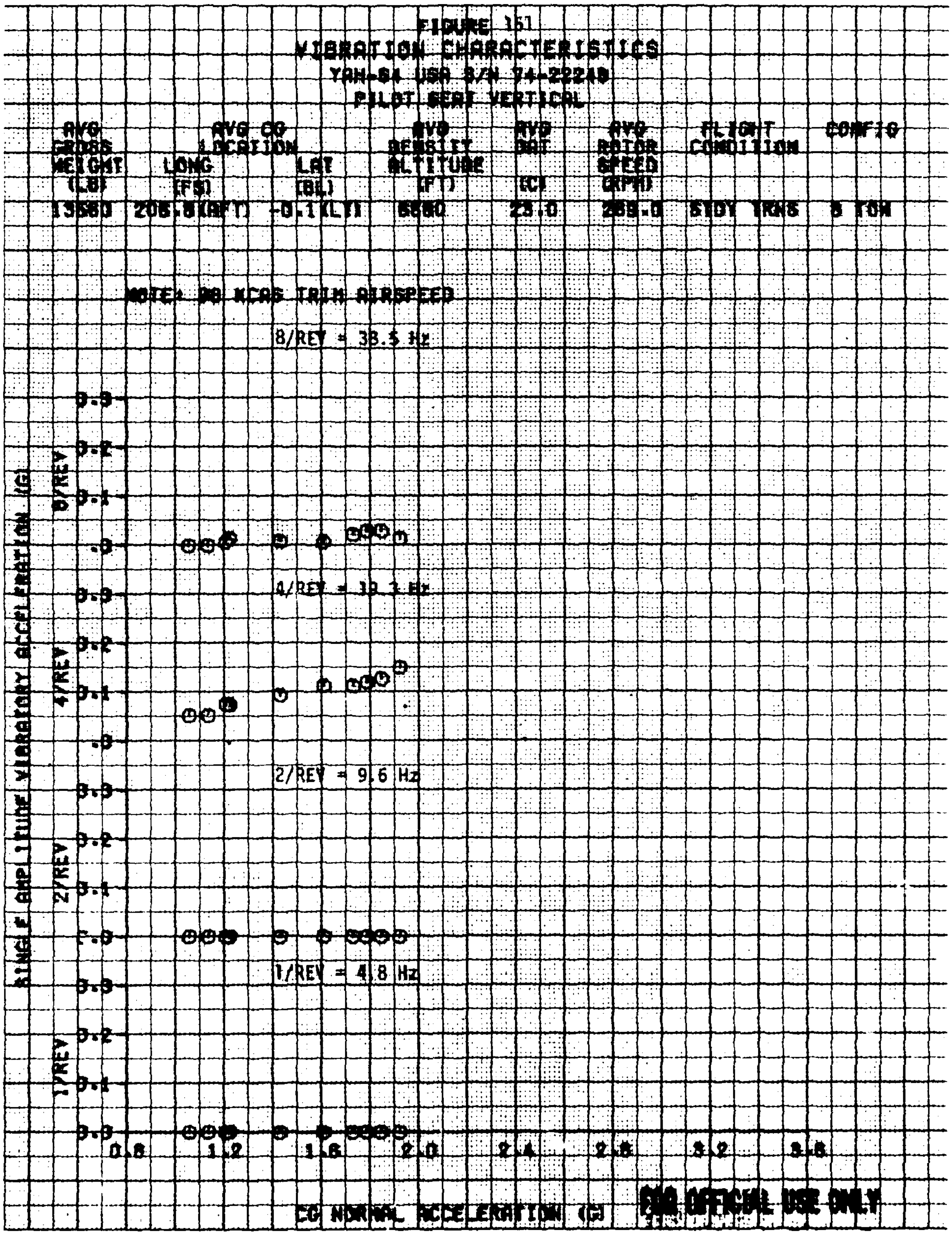


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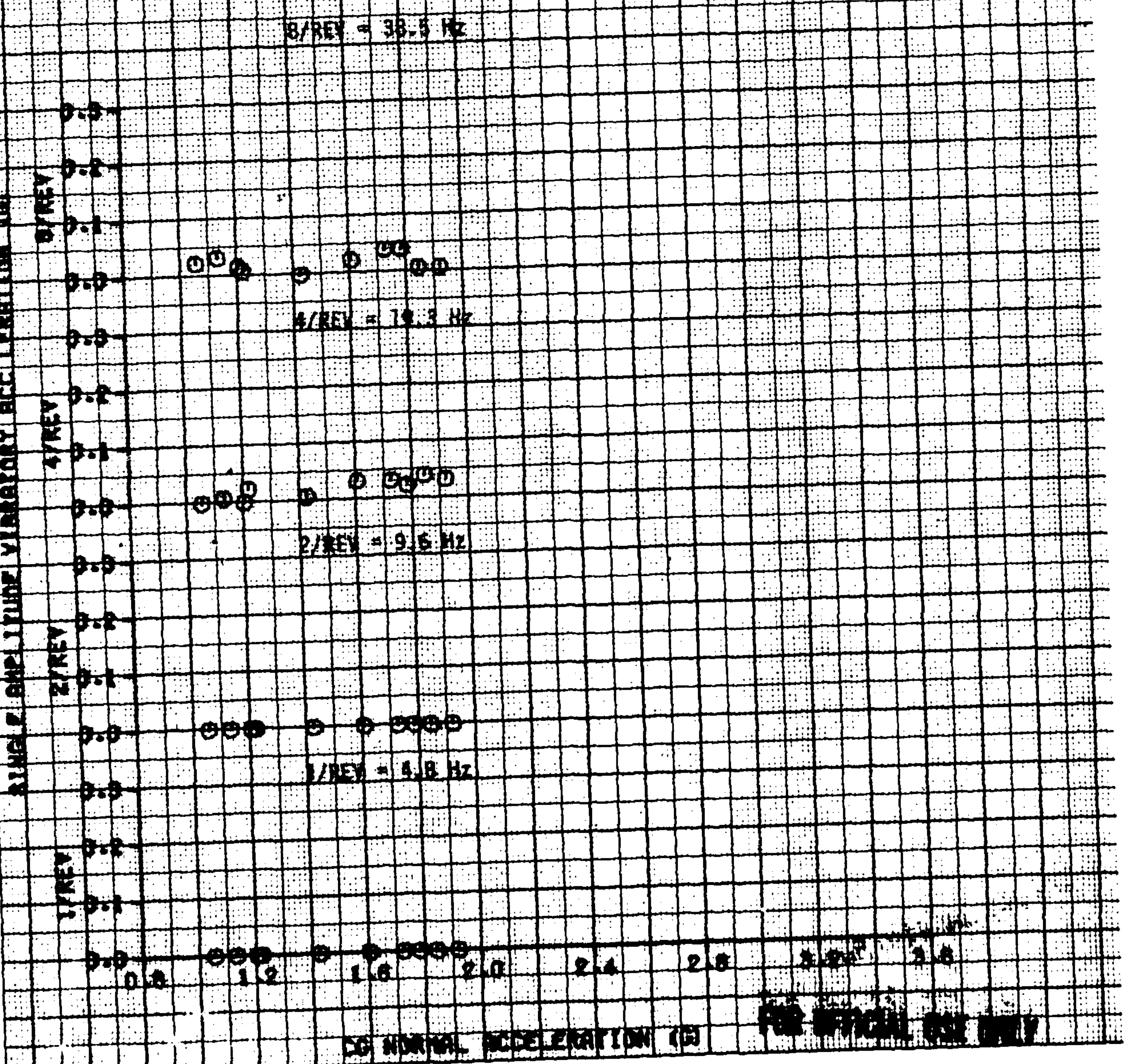
FIGURE 152
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22248
COPILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
13580	205.8(AFT)	-0.1(LT)	5850	23.0	289.0	STDY TRNS	& TON

NOTE: 98 KCAS TRIM AIRSPEED



FOR OFFICIAL USE ONLY

FIGURE 153
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 74-22249
C G VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
13560	205.8(AFT)	-0.1(LT)	[illegible]	25.0	289.0	STDY TRNS	[illegible]

NOTE: 98 KCAS TRIM AIRSPEED

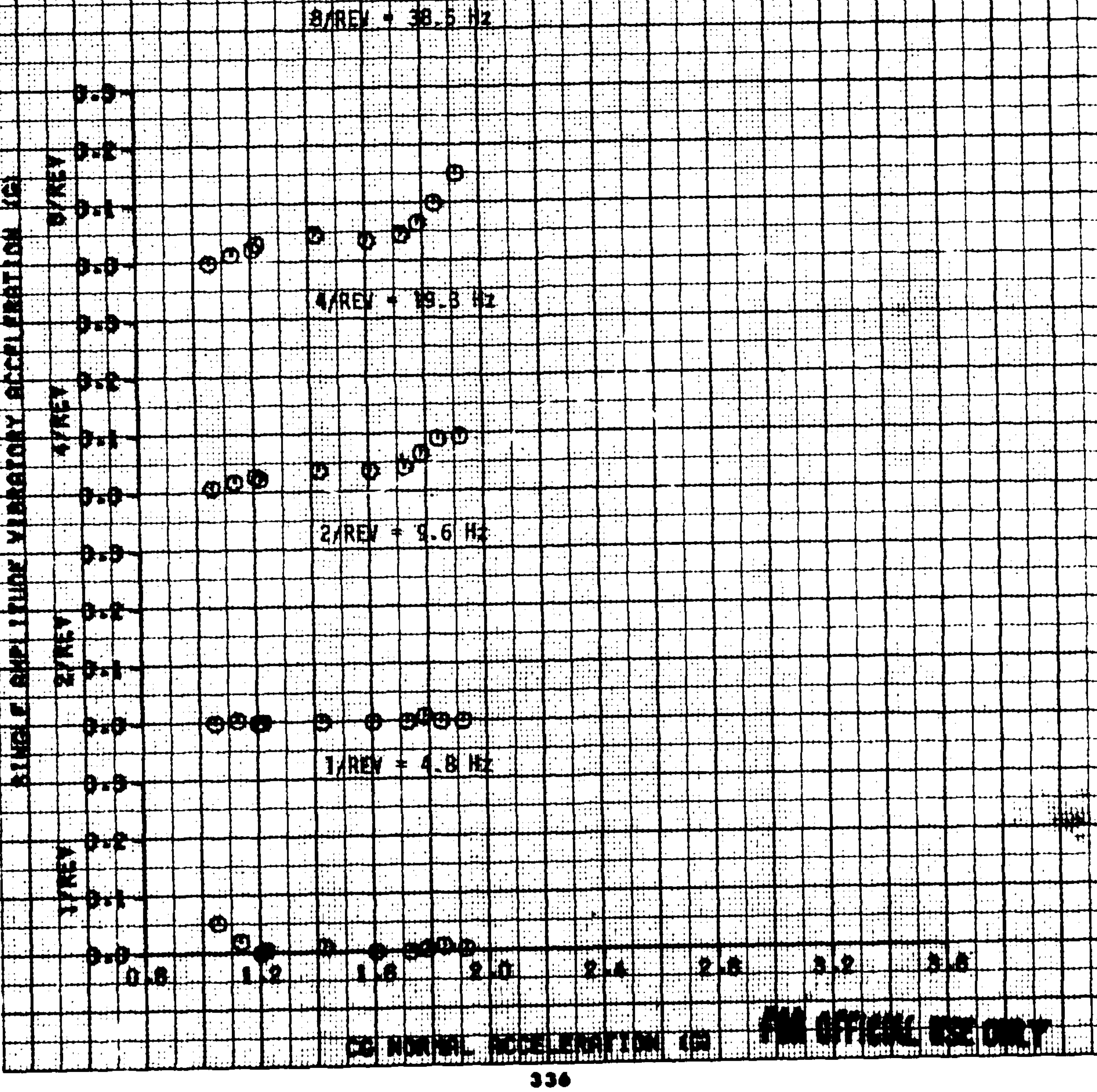


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FIGURE 154

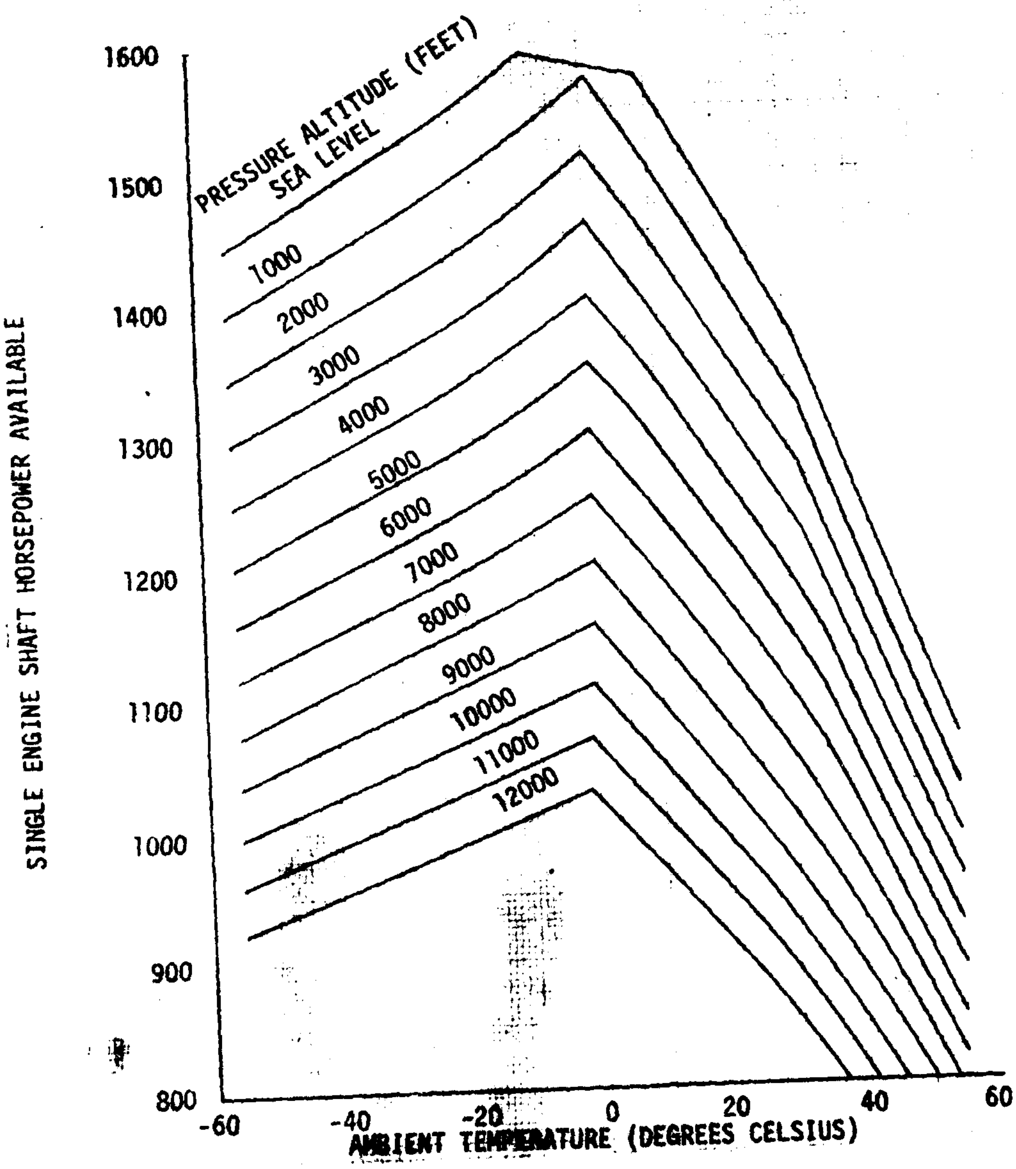
INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YAH-64 USA S/N 74-22249 T700-GE-700 LEFT ENGINE

19,979 OUTPUT SHAFT (289 ROTOR) RPM ZERO KNOTS TRUE AIRSPEED

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Engine inlet temperature rise = 0.9°C
2. Engine inlet pressure ratio = 1.004
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characterization as shown on figure



 FOR OFFICIAL USE ONLY

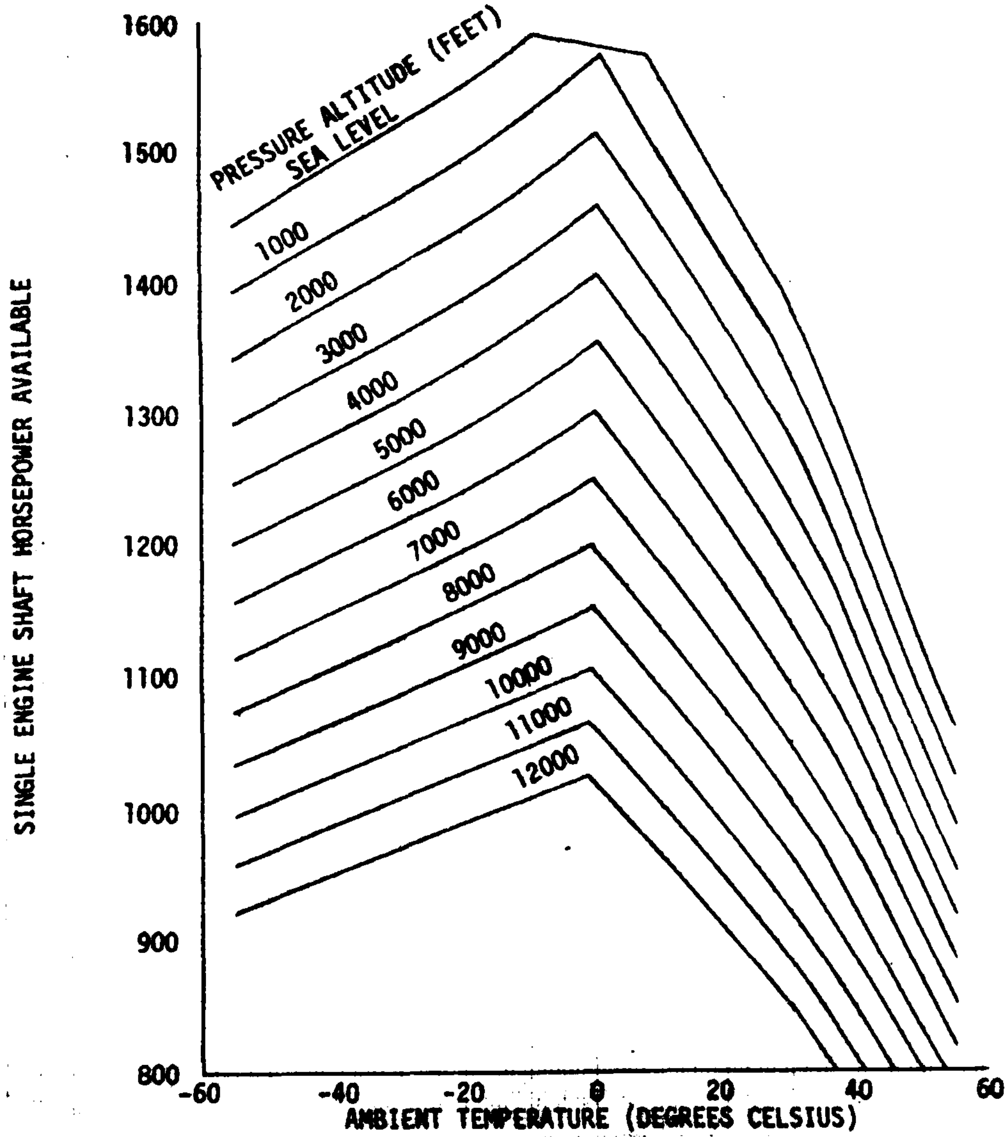
FIGURE 155

INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YAH-64 USA S/N 74-22249 T700-GE-700 RIGHT ENGINE

19,979 OUTPUT SHAFT (289 ROTOR) RPM ZERO KNOTS TRUE AIRSPEED

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:
1. Engine inlet temperature rise = 1.0°C
2. Engine inlet pressure ratio = 1.004
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characterization as shown on figure



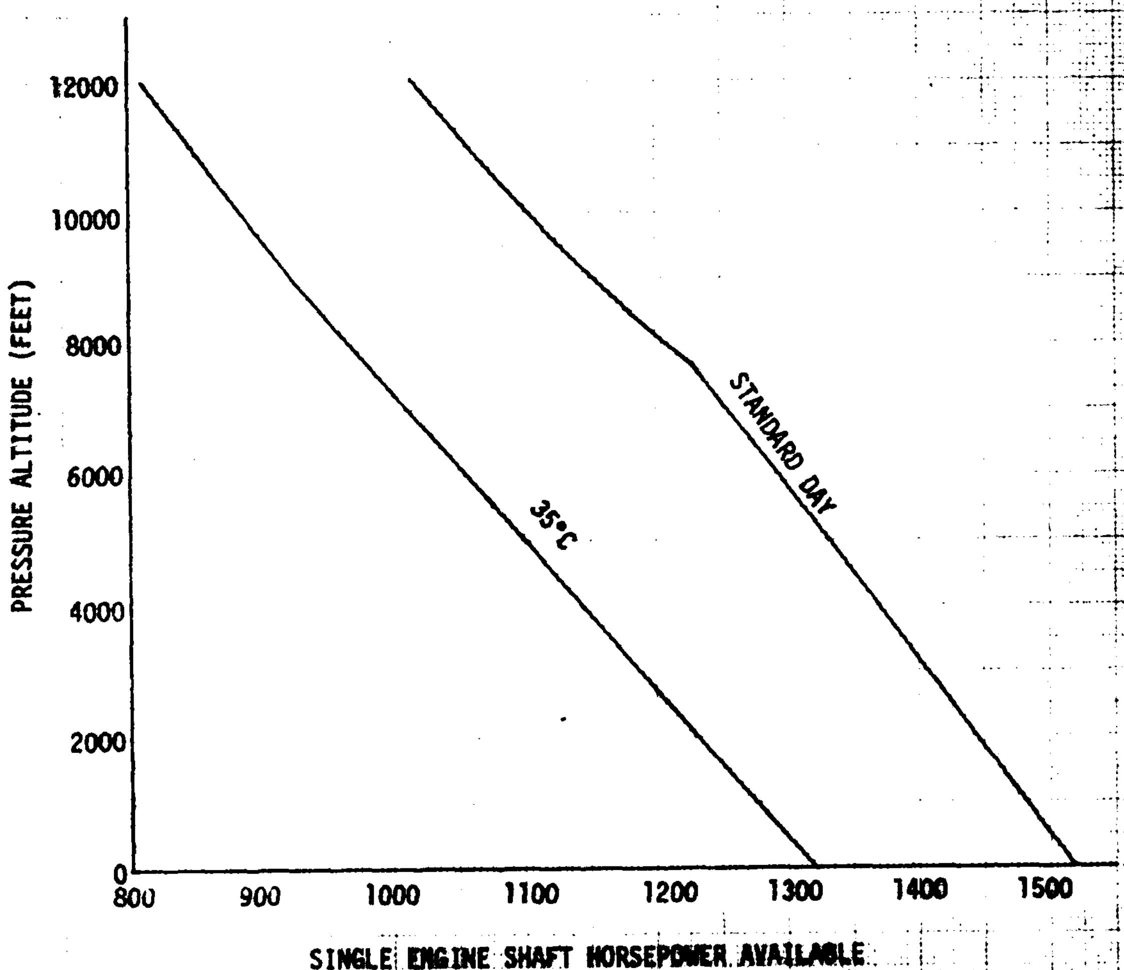
[illegible]

[illegible]

19,979 OUTPUT SHAFT ([illegible] ROTOR) RPM — ZERO KNOTS TRUE AIRSPEED

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-C2000, dated 2 Feb 73 corrected for the following installation conditions:

1. Engine inlet temperature rise = 0.9°C
2. Engine inlet pressure ratio = 1.004
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characterization as shown on figure



[illegible] POWER AVAILABLE
[illegible] ENGINE
[illegible] AIRSPEED

NOTE: Based on [illegible] Specification [illegible] dated 2 [illegible] 73 [illegible] for the following installation conditions:

1. Engine inlet temperature rise = 1.0°C
2. Engine inlet pressure ratio = 1.004°C
3. [illegible] bleed air = zero
4. [illegible] off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characterization as shown on figure

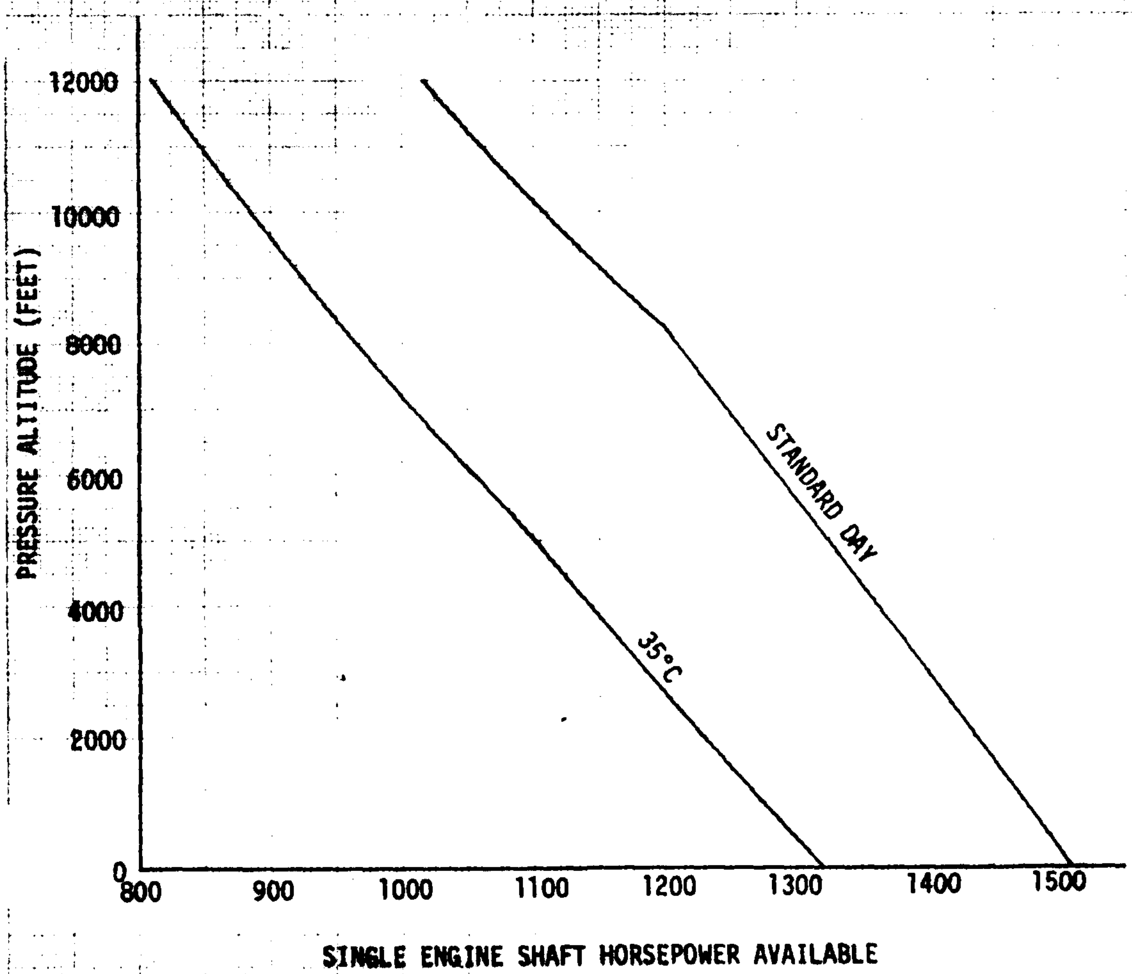


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FIGURE 158

INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE VERSUS AIRSPEED

YAH-64 USA S/N 74-222249 T700-GE-700 LEFT ENGINE
19,979 OUTPUT SHAFT (289 ROTOR) RPM

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:
1. Inlet characteristics obtained from figures , and
2. Engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb

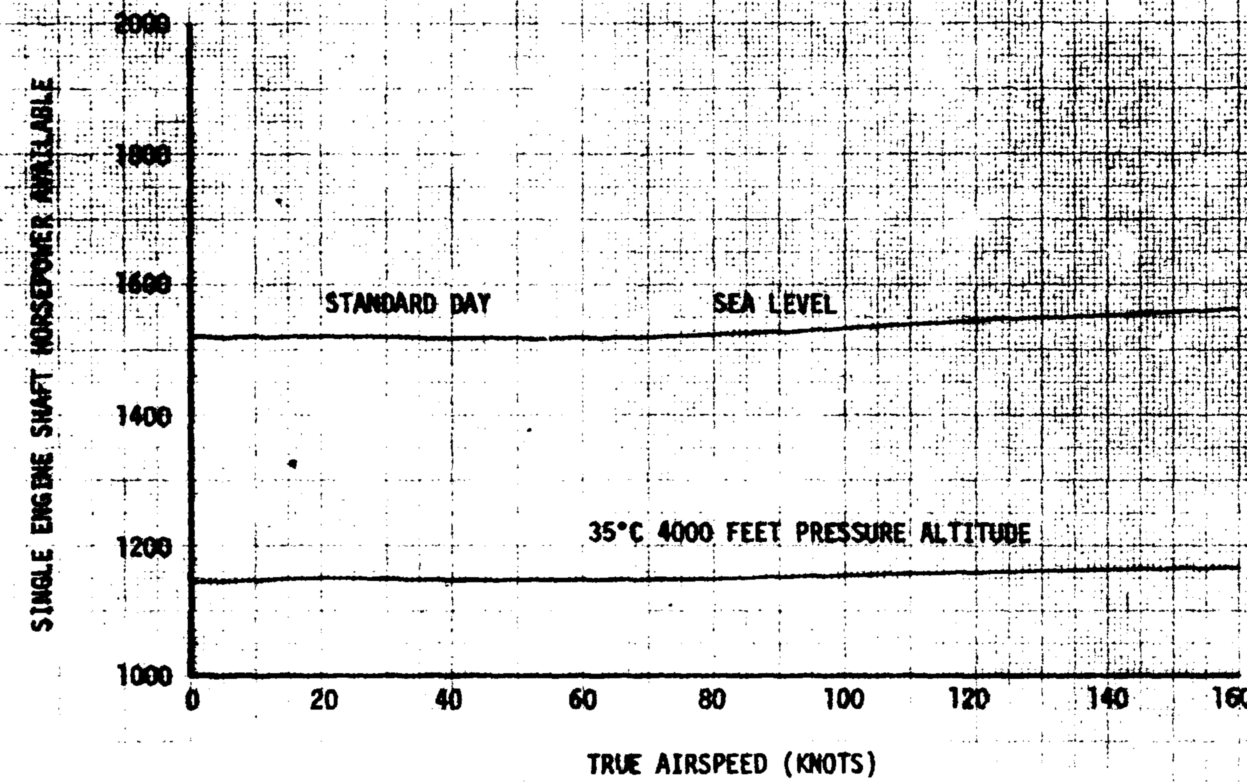

FIGURE 159

POWER AVAILABLE VERSUS AIRSPEED

YAH-64 USA S/N 74-22249 T700-GE-700 BOTH ENGINES

19,979 OUTPUT SHAFT (289 ROTOR) RPM AMBIENT TEMPERATURE 35°C PRESSURE ALTITUDE (FEET) 4000

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Inlet characteristics obtained from figures , and .
2. Engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb
5. G.E. Referee exhaust system characterization

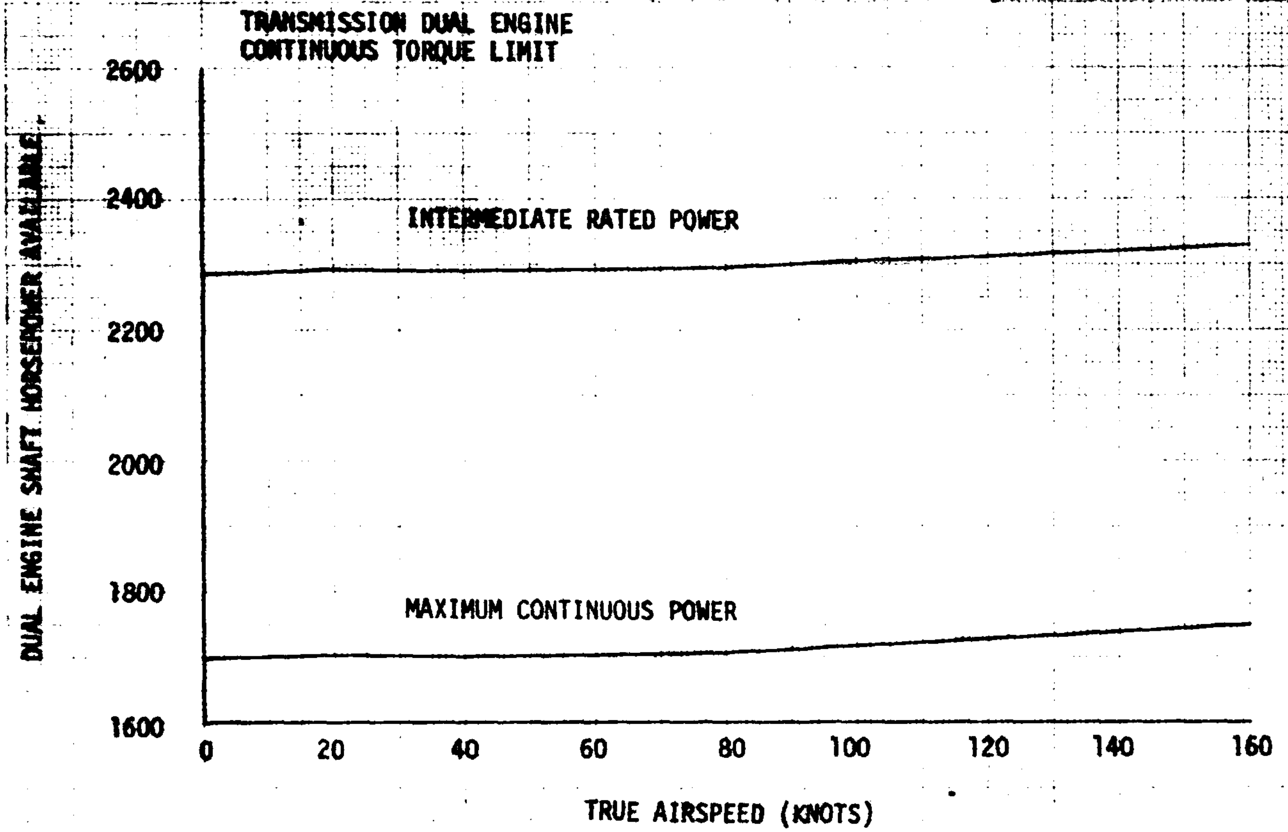

FIGURE 160

POWER AVAILABLE VERSUS AIRSPEED

YAH-64 USA S/N 74-22249 T700-GE-700 BOTH ENGINES

1?,979 OUTPUT SHAFT (28? ROTOR) RPM AMBIENT TEMPERATURE 15 C PRESSURE ALTITUDE - SEA LEVEL

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Inlet characteristics obtained from figures , and
2. Engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb

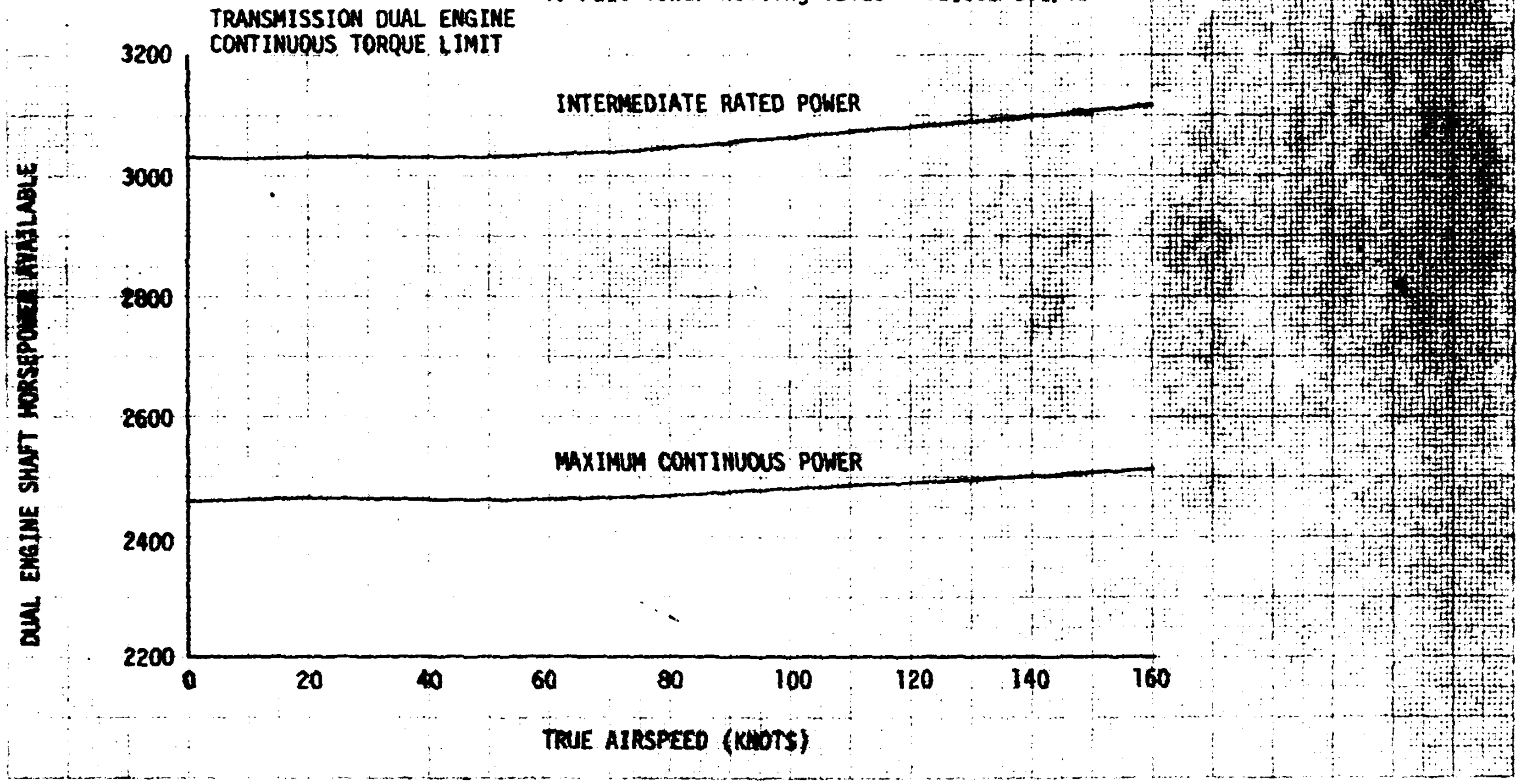

FIGURE 161

DUAL ENGINE SPECIFICATION FUEL FLOW

YAH-64 USA S/N 74-22249 T700-GE-700 BOTH ENGINES

19,979 OUTPUT SHAFT (289 ROTOR) RPM STANDARD DAY PRESSURE ALTITUDE = SEA LEVEL

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73- corrected for the following installation conditions:

1. Inlet and exhaust characteristics obtained from figure
2. Customer bleed air and engine anti-ice off
3. Fuel lower heating value = 18,300 BTU/lb

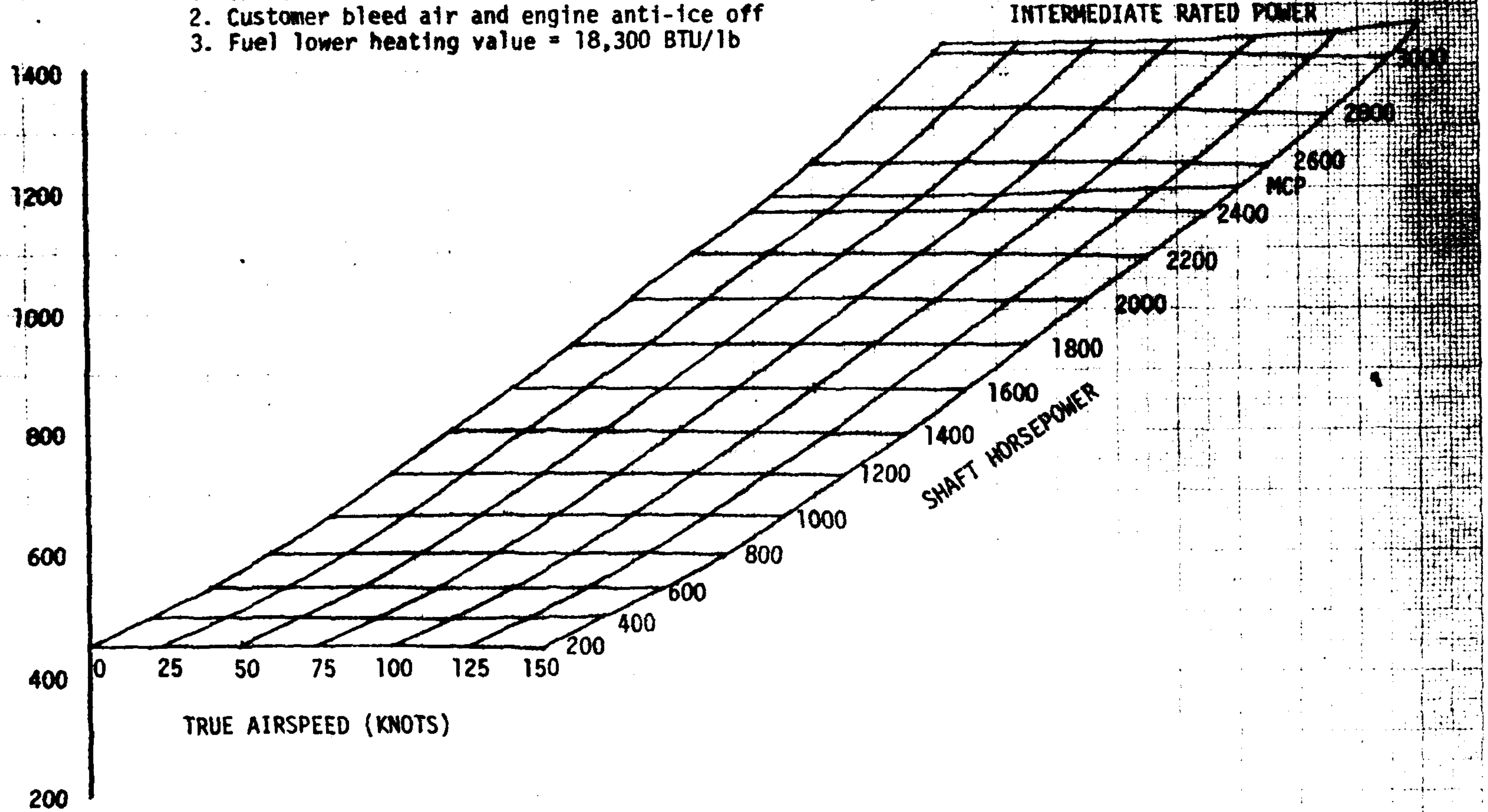

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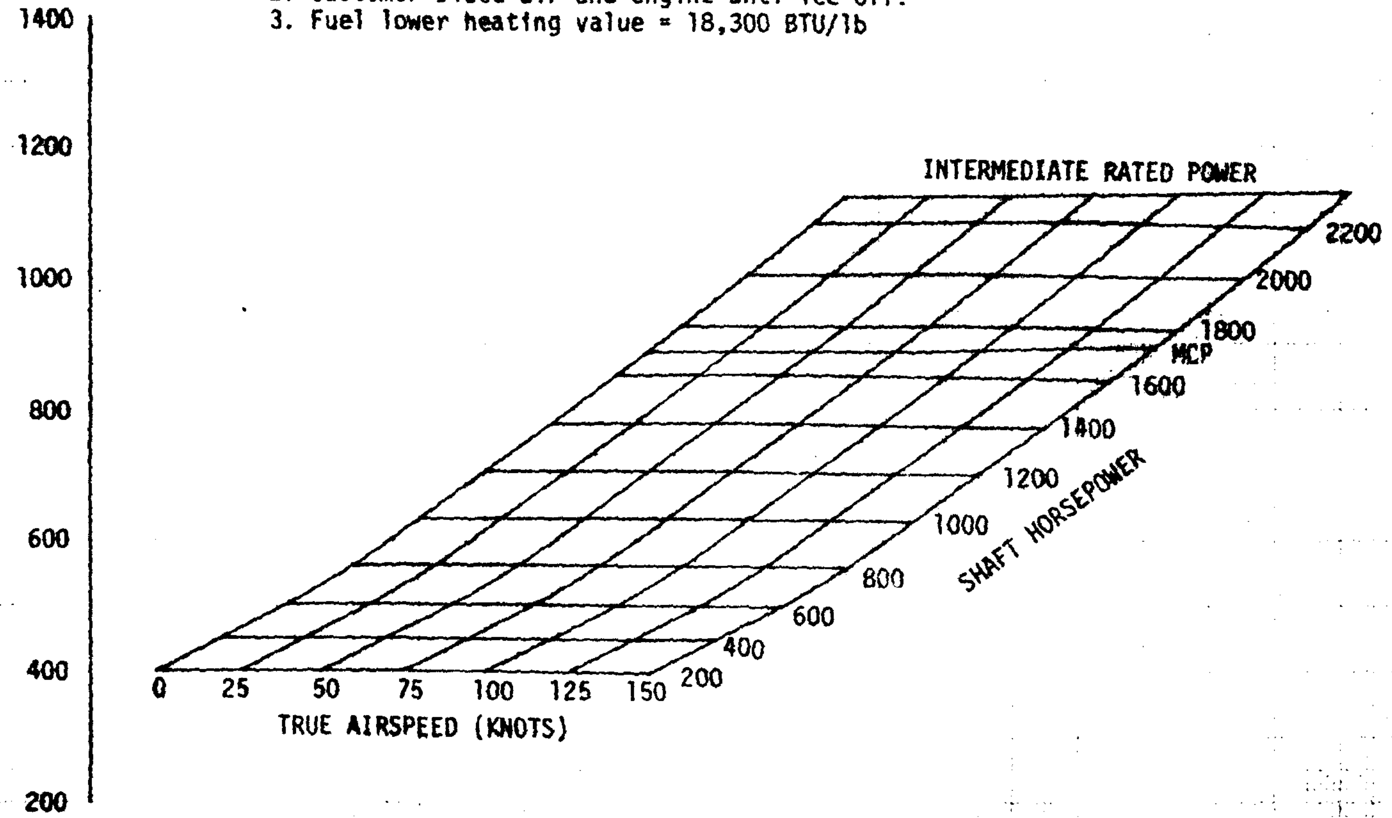
FIGURE 162

DUAL ENGINE SPECIFICATION FUEL FLOW

YUH-64 USA S/N 74-22249 T700-GE-700 BOTH ENGINES

19,979 OUTPUT SHAFT (289 ROTOR) RPM AMBIENT TEMPERATURE = 35°C PRESSURE ALTITUDE = 4000 FEET

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:
1. Inlet and exhaust characteristics obtained from figures
2. Customer bleed air and engine anti-ice off.
3. Fuel lower heating value = 18,300 BTU/lb



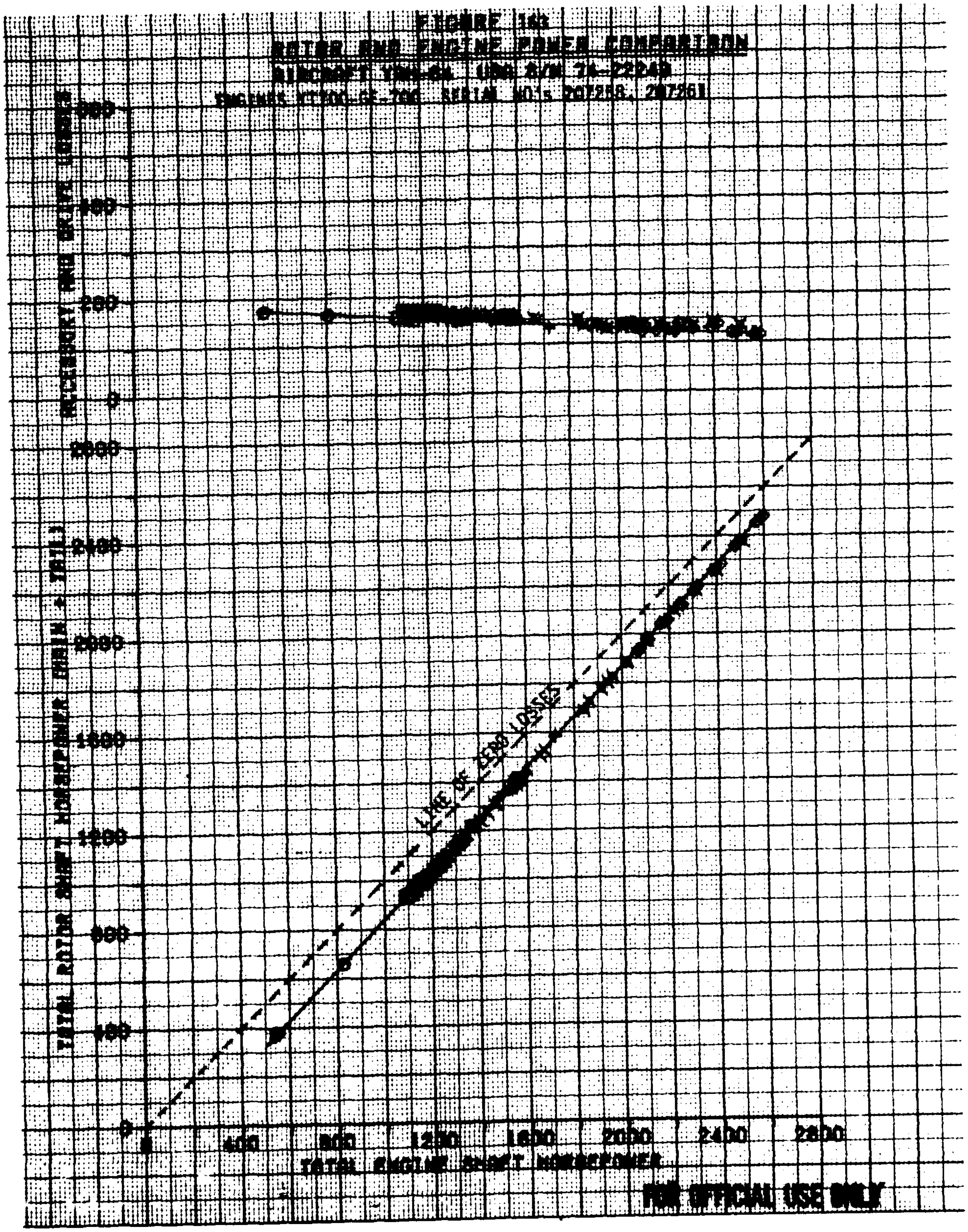

ROTOR AND ENGINE POWER COMPARISON
LINE OF ZERO LOSSES
TOTAL ENGINE SHAFT HORSEPOWER
0
400
800
1200
1600
2000
2400
2800
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FIGURE 164

ENGINE INLET TEMPERATURE RISE

AIRCRAFT YAH-64 USA S/N 74-22249 YT700-GE-700 ENGINES S/N S 207258, 207261

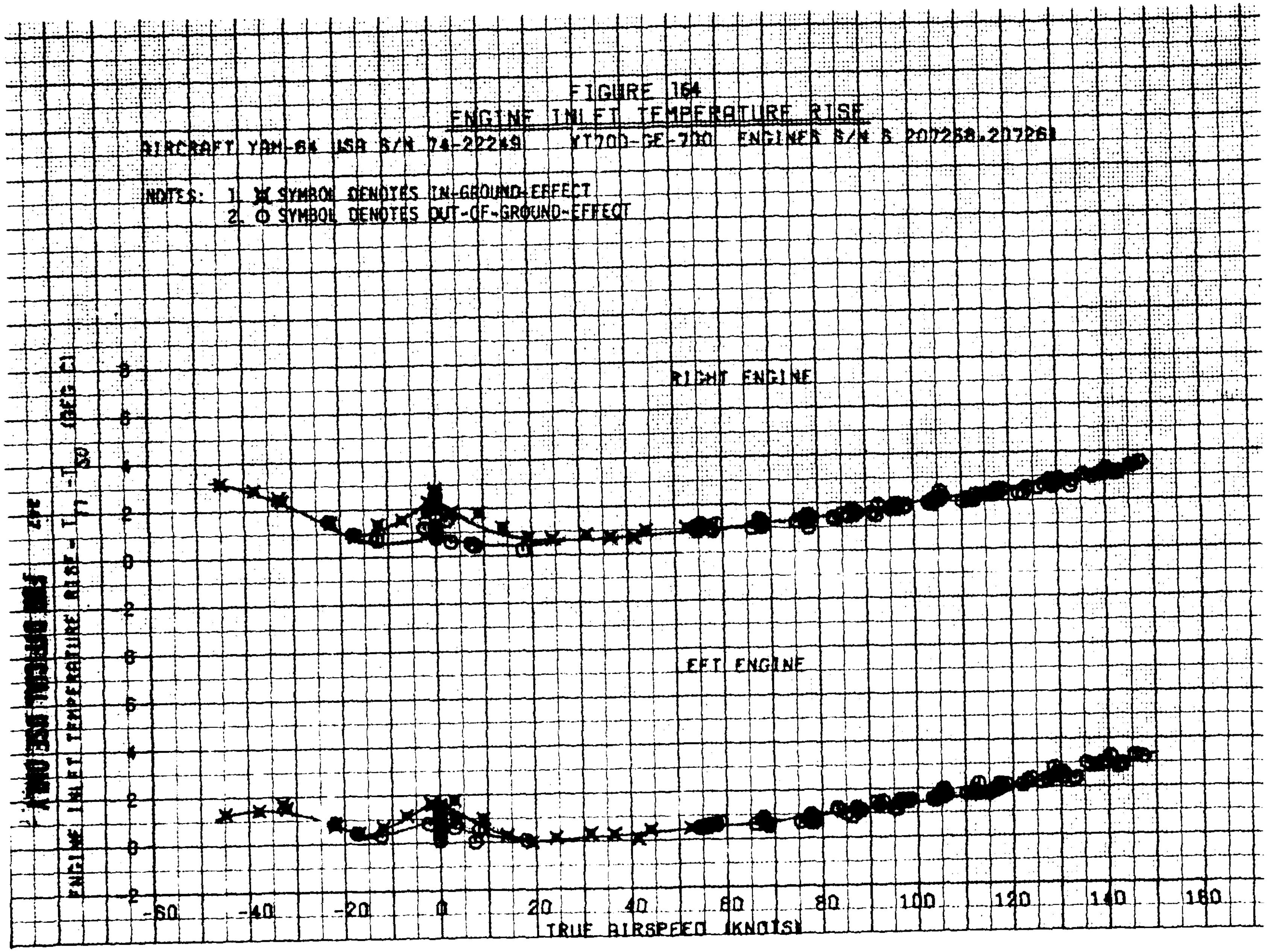


FIGURE 165
ENGINE INLET PRESSURE RATIOS
AIRCRAFT YAH-64 USA S/N 74-22249 YT700-GE-700 ENGINES S/N S 207260,207261

NOTES: 1. ✕ SYMBOL DENOTES IN-GROUND-EFFECT
2. O SYMBOL DENOTES OUT-OF-GROUND-EFFECT

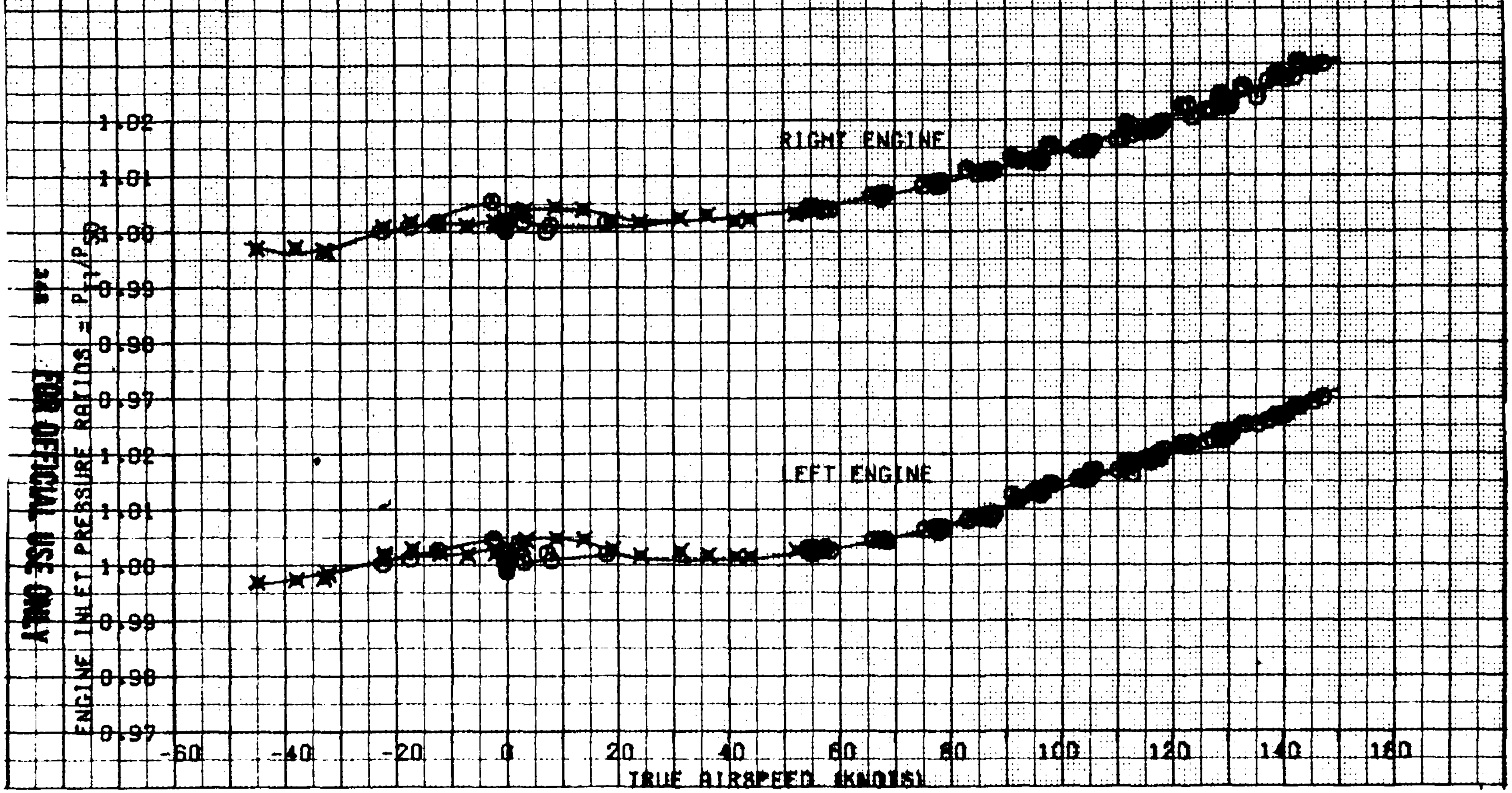


FIGURE 166

ENGINE INLET TEMPERATURE RISE IN SIDEWARD FLIGHT

AIRCRAFT YAH-64 USA S/N 74-22249 YT700-GE-700 ENGINES S/N'S 207250,207281

NOTES: 1. ☒ SYMBOL DENOTES IN-GROUND-EFFECT
2. △ SYMBOL DENOTES OUT-OF-GROUND-EFFECT

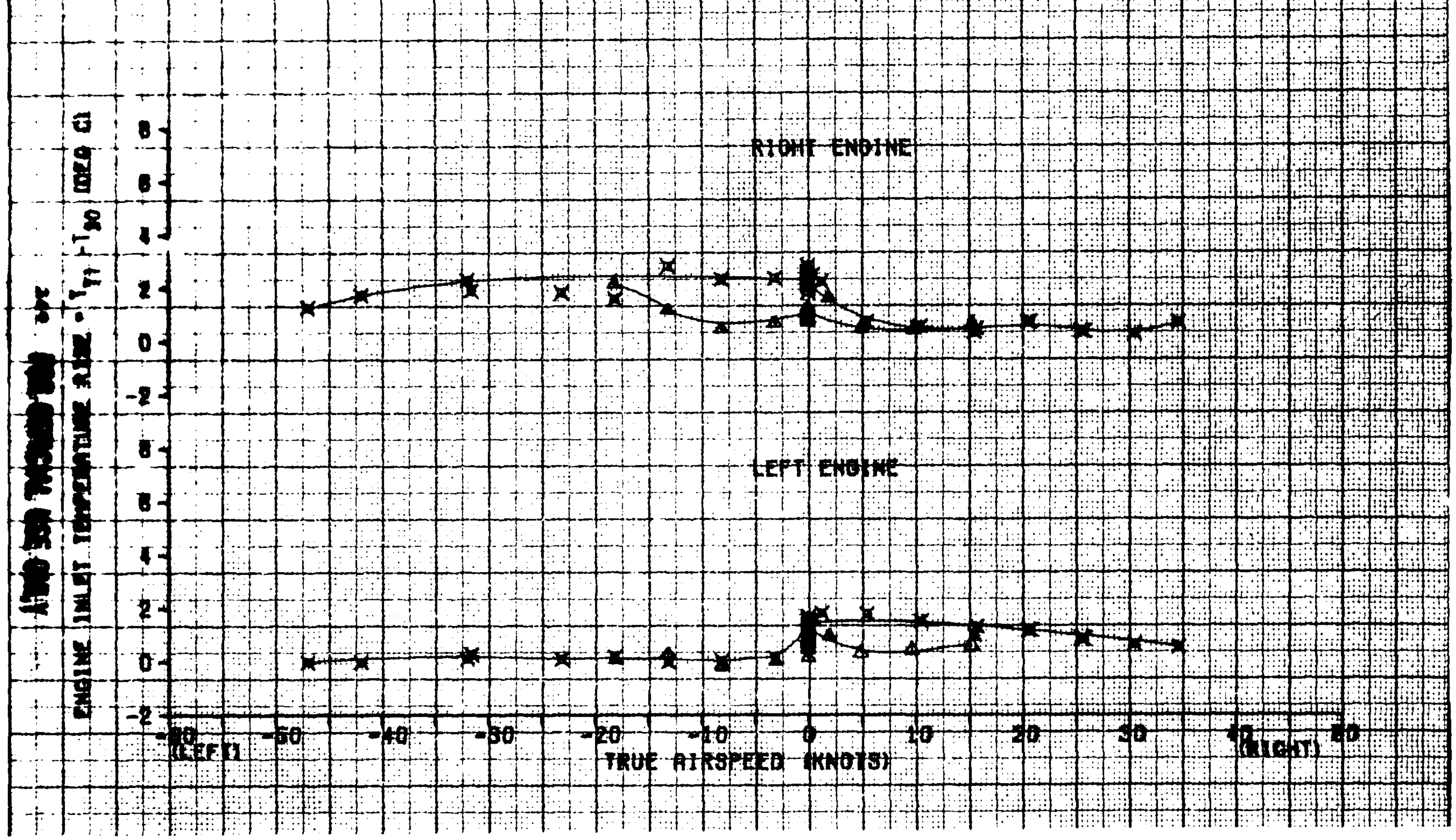


FIGURE 167

ENGINE INLET PRESSURE RATIOS IN SIDEWARD FLIGHT

AIRCRAFT YAH-64 USA S/N 74-22249   YT700-GE-700 ENGINES S/N'S 207258,207261

NOTES: 1. ⌧ SYMBOL DENOTES IN-GROUND-EFFECT
2. Δ SYMBOL DENOTES OUT-OF-GROUND-EFFECT

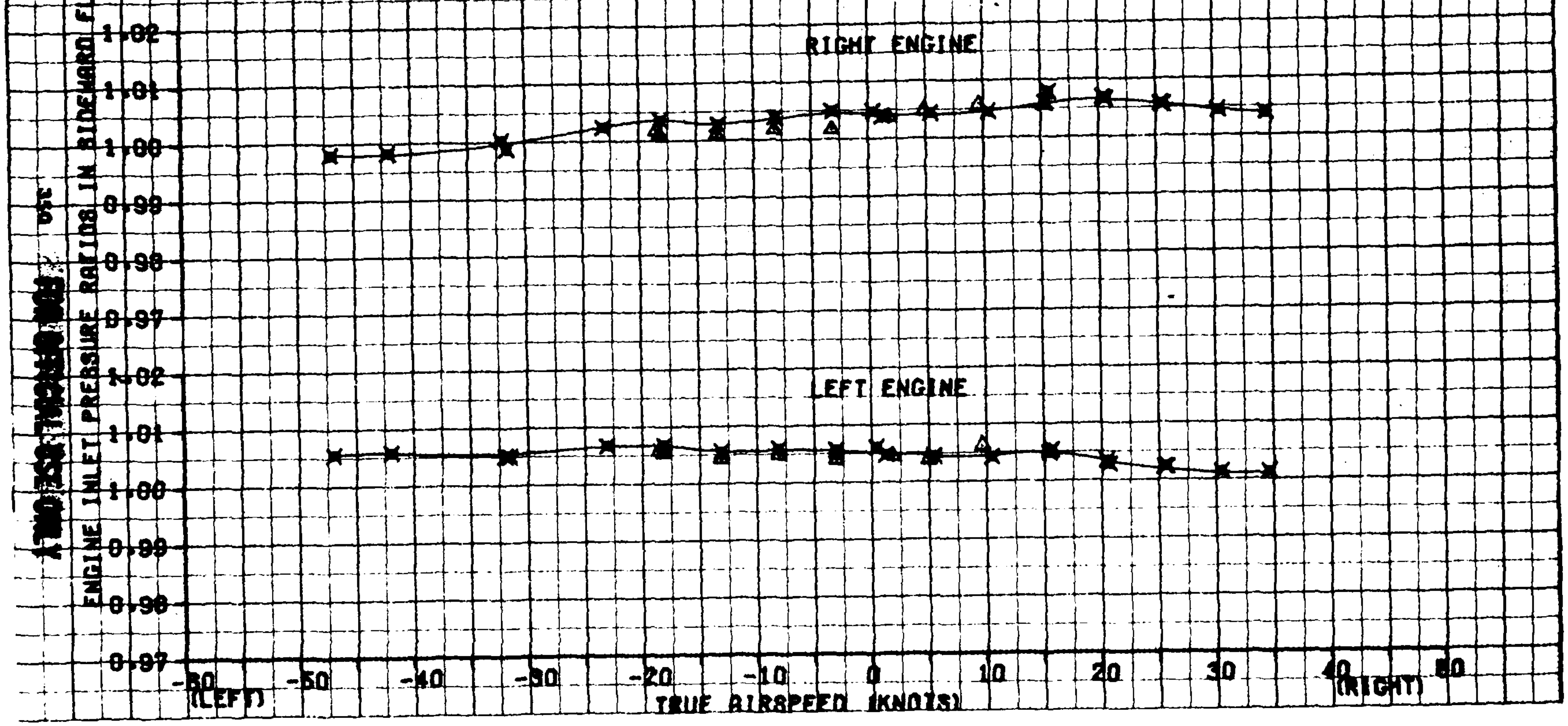


FIGURE 168

ENGINE INLET TEMPERATURE VARIATION WITH SIDESLIP

AIRCRAFT YAH-64 USA S/N 74-22249   YT700-GE-700 ENGINES S/N S 207258,207261

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CENTER OF GRAVITY (IN.)	AVG PRESSURE ALTITUDE (FT)	AVG OUTSIDE AIR TEMP (°C)	AVG ROTOR SPEED (RPM)	AVG TRUE AIRSPEED (KTAS)
□	15090	205.8 (AFT)	3820	17	290	66
○	14560	205.9 (AFT)	4480	16	289	86
◇	14900	205.9 (AFT)	4100	17	289	101
△	14700	205.9 (AFT)	4240	16	289	137

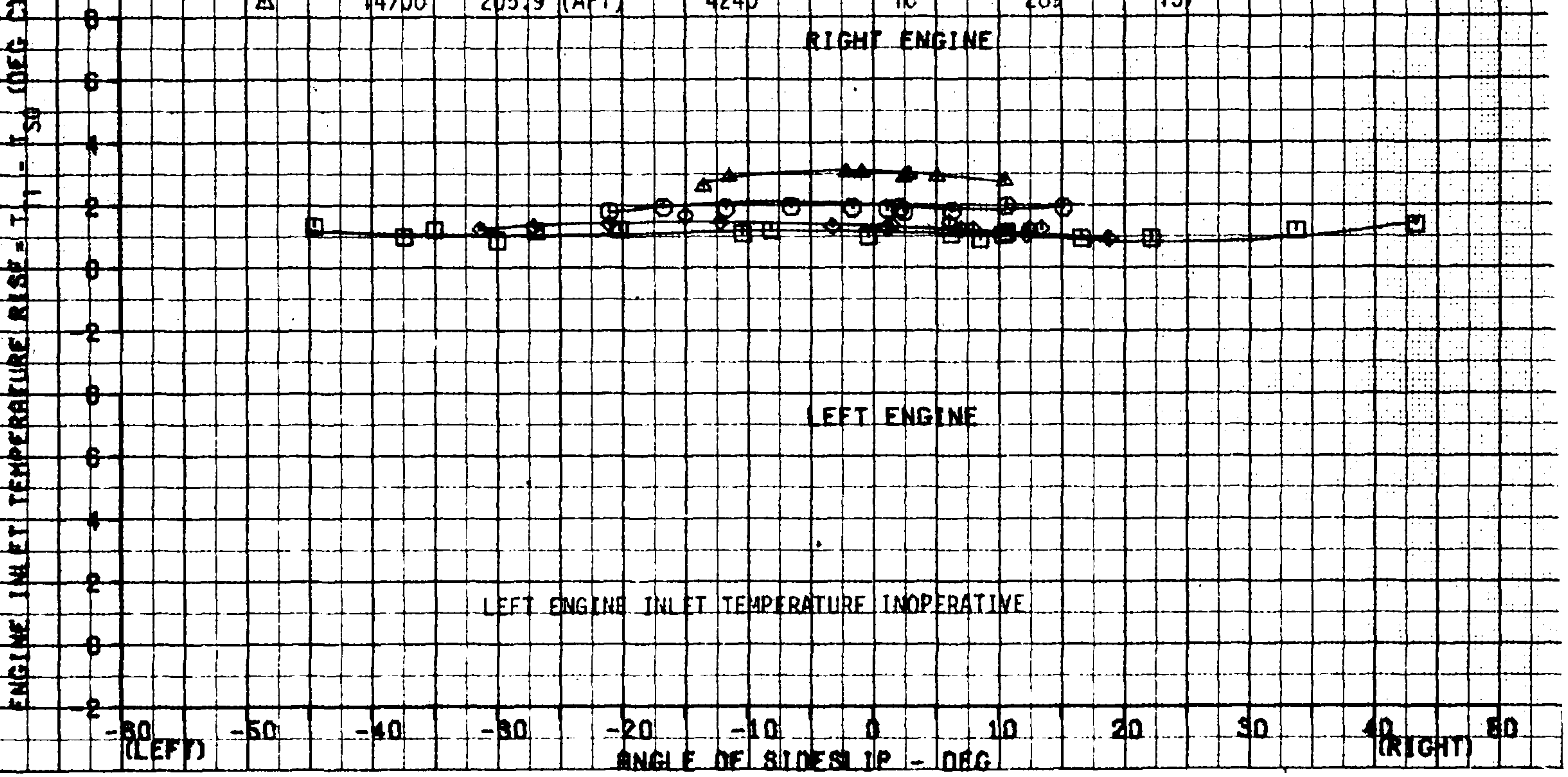


FIGURE 169

ENGINE INLET PRESSURE RATIO VARIATION WITH SIDESLIP

AIRCRAFT YAH-64 USA S/N 74-22249 YT700-GE-700 ENGINES S/N S 207258,207261

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CENTER OF GRAVITY (IN.)	AVG PRESSURE ALTITUDE (FT)	AVG OUTSIDE AIR TEMP (°C)	AVG ROTOR SPEED (RPM)	AVG TRUE AIRSPEED (KTAS)
□	15090	205.8 (AFT)	3820	17	290	66
○	14560	205.9 (AFT)	4480	16	289	85
◇	14900	205.9 (AFT)	4100	17	289	101
△	14700	205.9 (AFT)	4240	16	289	137

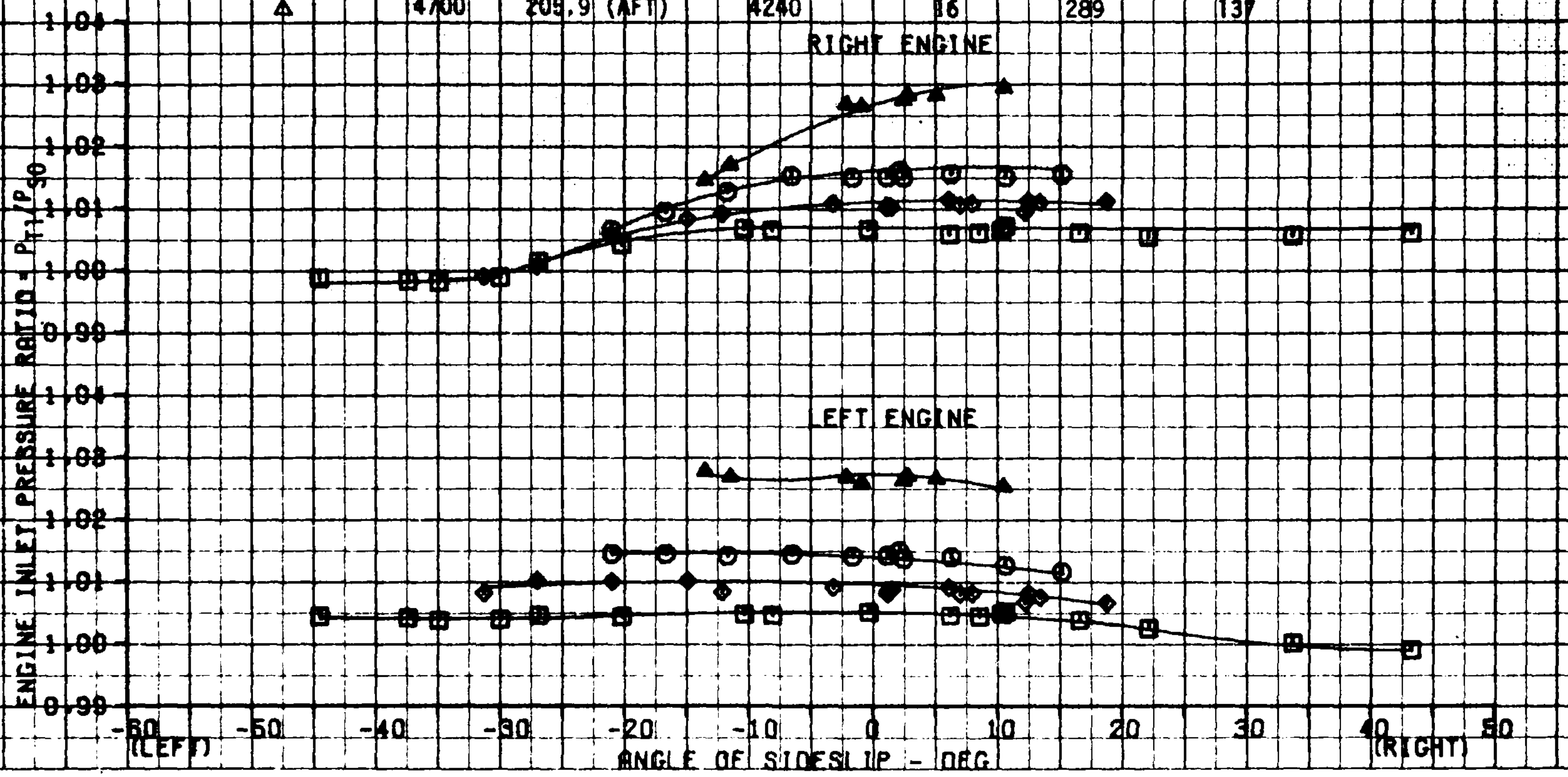


FIGURE 170
ENGINE INLET TEMPERATURE RISE IN VERTICAL CLIMB
AIRCRAFT YAH-64 USA S/N 74-22249   YT700-GE-700 ENGINES S/N'S 207256,207261

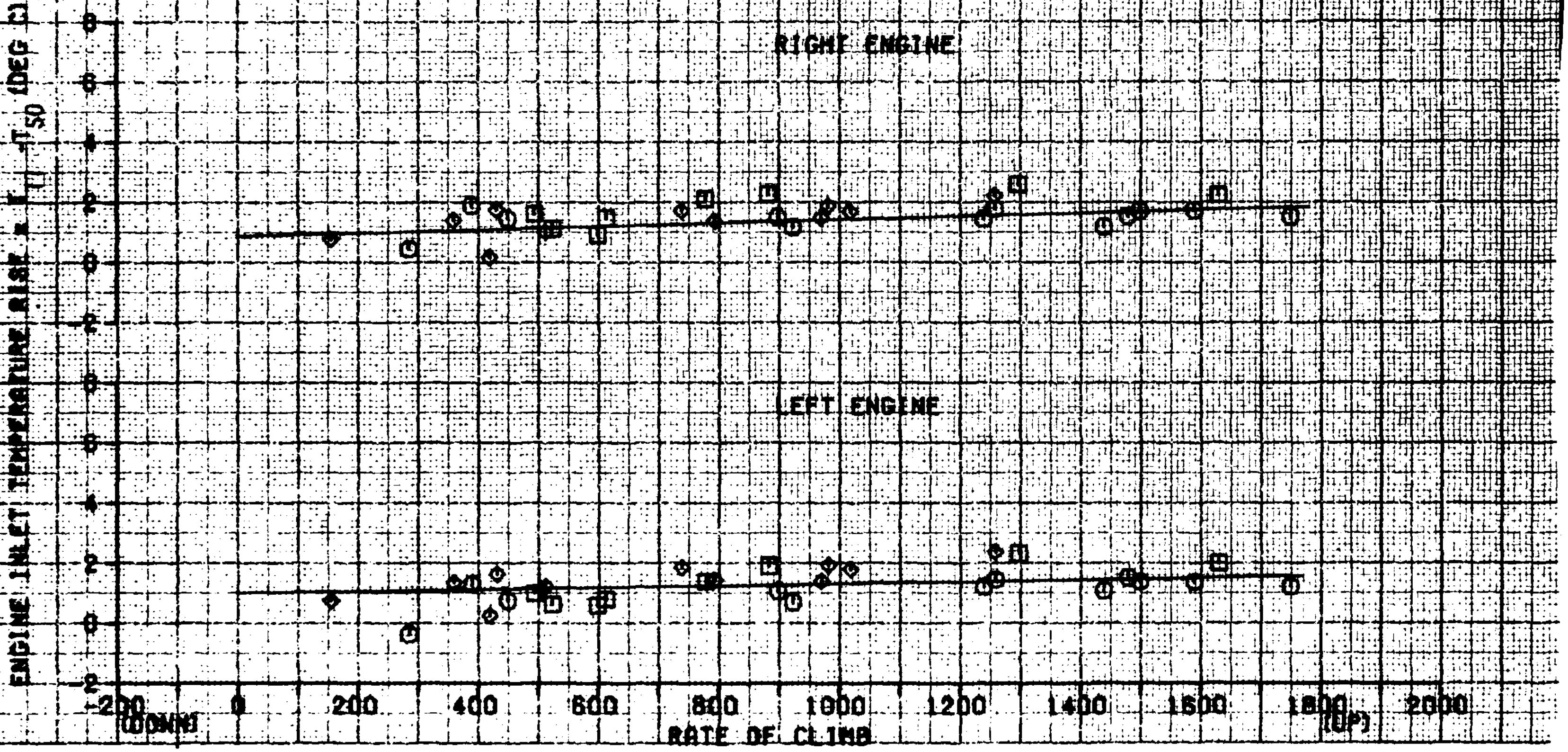
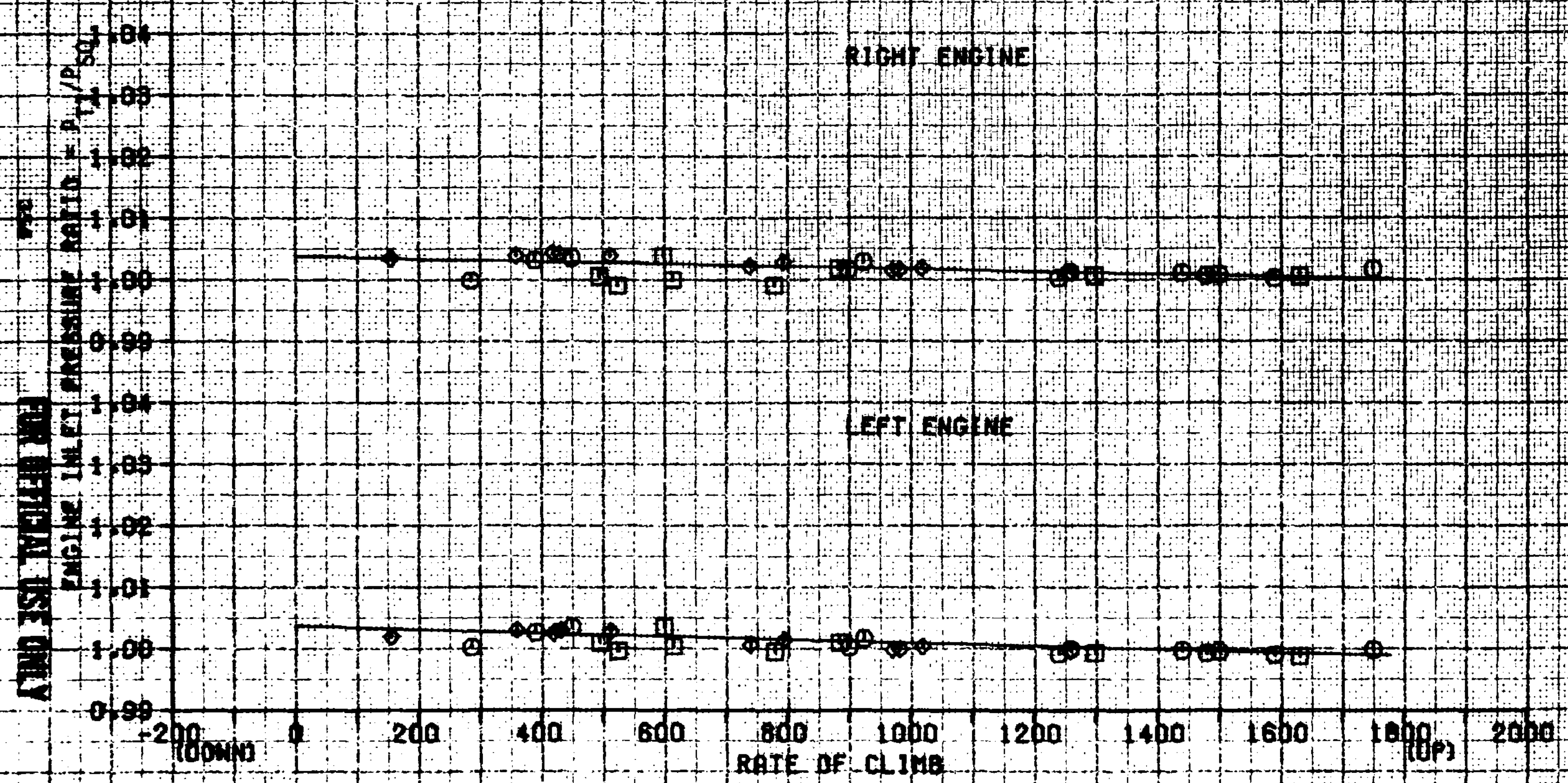


FIGURE 171
ENGINE INLET PRESSURE RATIO IN VERTICAL CLIMB
AIRCRAFT YAH-64 USA S/N 74-22249 YT700-GE-700 ENGINES S/N'S 207258,207261



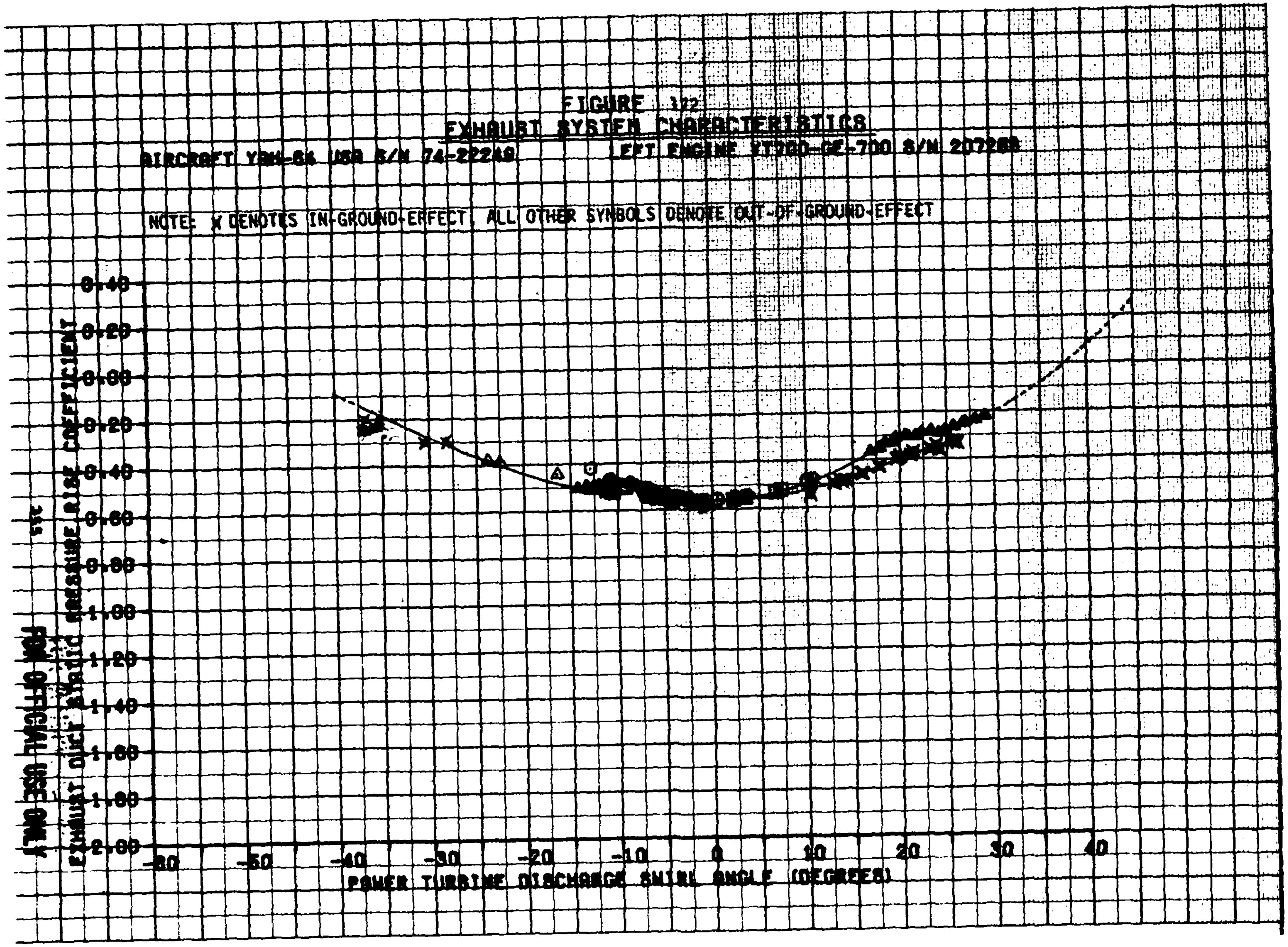


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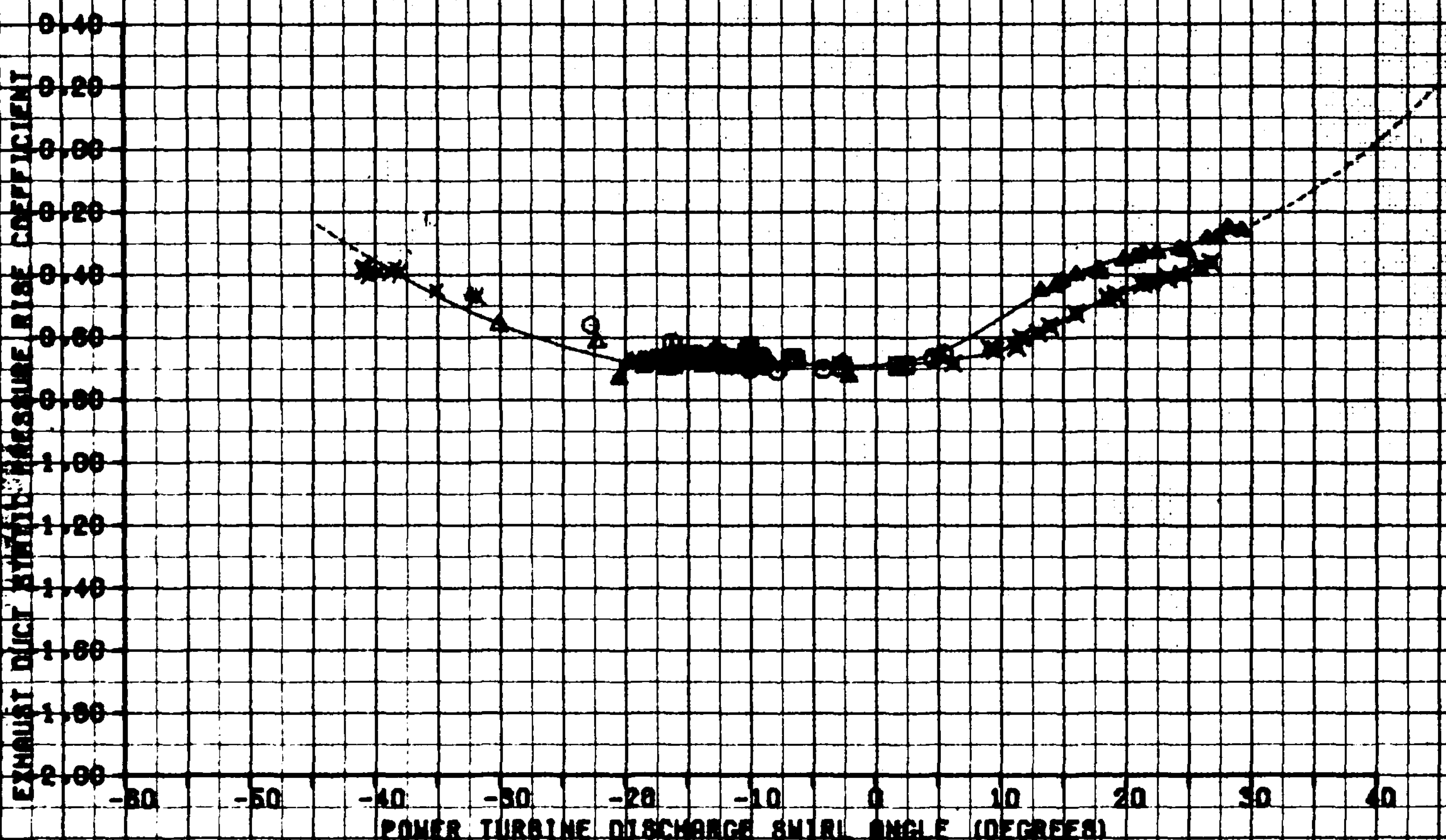
FIGURE 173

EXHAUST SYSTEM CHARACTERISTICS

AIRCRAFT YAH-64 USA S/N 74-22248   RIGHT ENGINE YT700-GE-700 S/N 207250

NOTE: x DENOTES IN-GROUND-EFFECT; ALL OTHER SYMBOLS DENOTE OUT-OF-GROUND-EFFECT



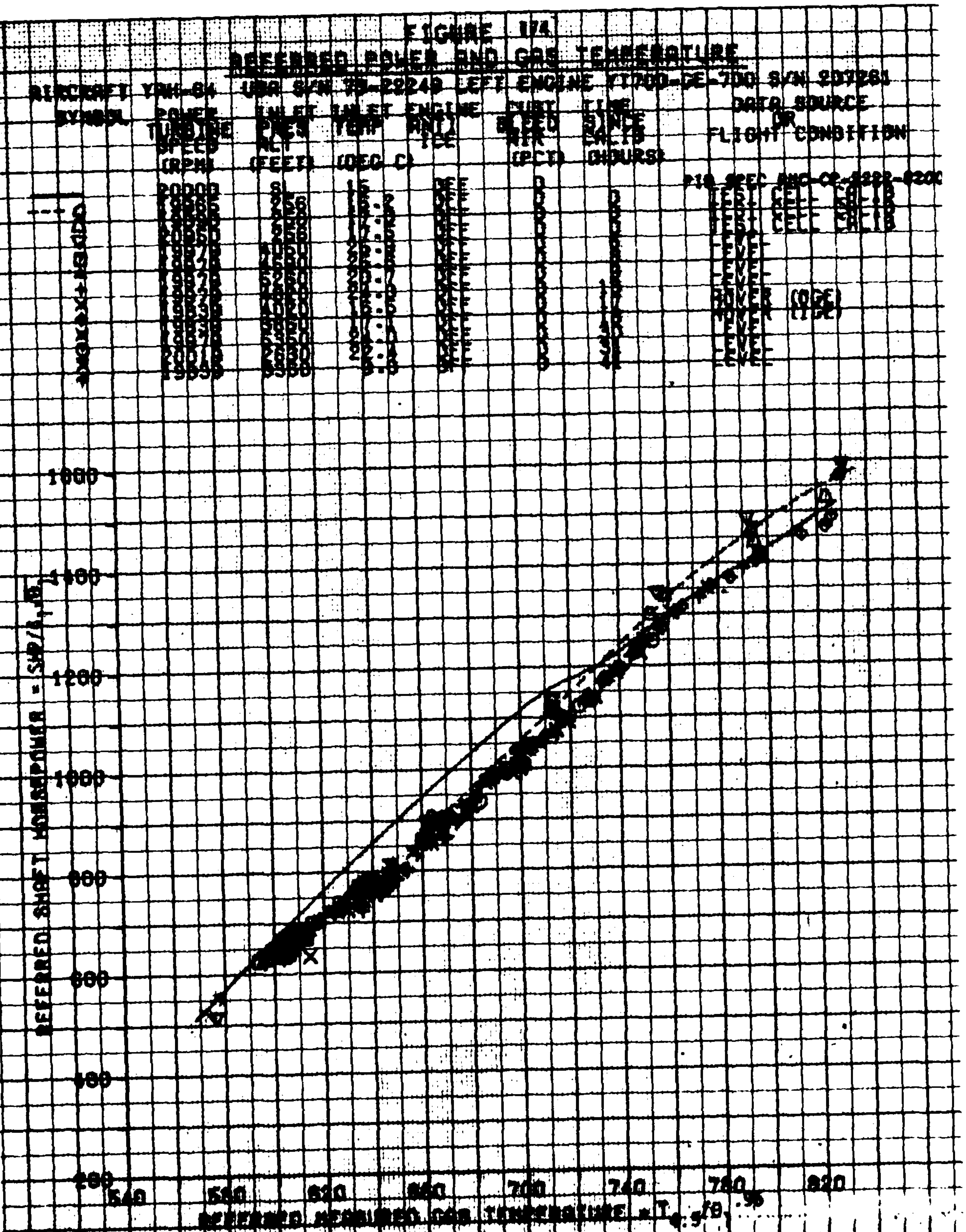


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FIGURE 175

REFERRED POWER AND GAS TEMPERATURE

AIRCRAFT YAH-64 USA S/N 73-22249 RIGHT ENGINE YT700-GE-700 S/N 207258

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20000	SL	15	OFF	0		PID SPEC AMC-CP-2222-02000
- - - ○	20080	180	18.0	OFF	0	0	TEST CELL CALIB
△	[illegible]	180	18.2	OFF	0	0	TEST CELL CALIB
▽	20010	180	17.5	OFF	0	0	TEST CELL CALIB
□	19978	4050	25.8	OFF	0	8	LEVEL
◇	19978	4590	25.5	OFF	0	8	LEVEL
+	19978	5260	20.7	OFF	0	8	LEVEL
X	[illegible]	[illegible]	[illegible]	OFF	0	13	LEVEL
◇	19839	4020	16.5	OFF	0	17	HOVER (OGE)
X	19839	3860	17.1	OFF	0	18	HOVER (IGE)
⊙	19878	5350	[illegible]	OFF	0	30	LEVEL
⊠	20018	2630	22.4	OFF	0	31	LEVEL
✦	19839	6360	9.0	OFF	0	41	LEVEL

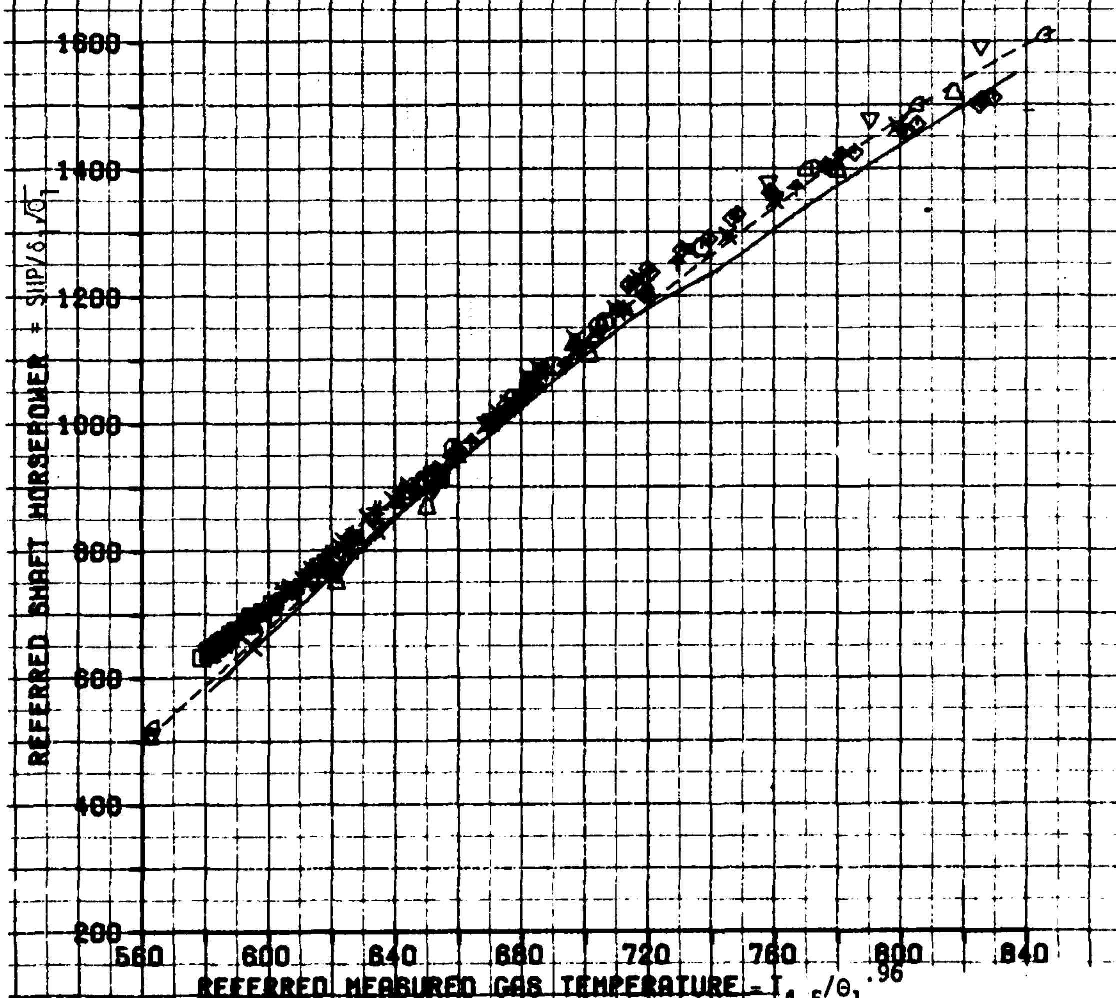
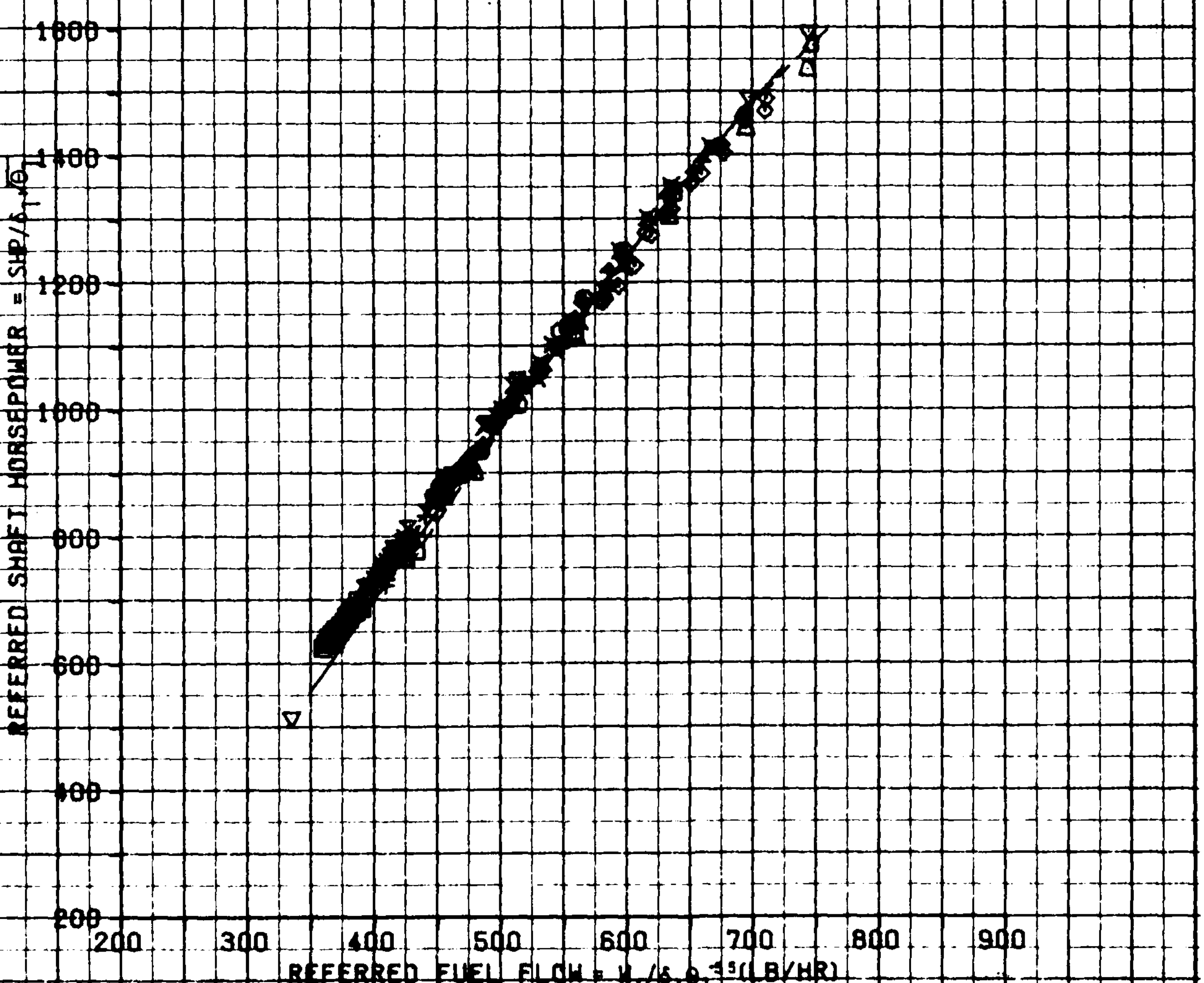


FIGURE 176
REFERRED POWER AND FUEL FLOW

AIRCRAFT YAH-64 USA S/N 73-22249 LEFT ENGINE YT700-GE-700 S/N 207261

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20000	SL	15	OFF	0		PIO SPEC ANC-CP-2222-02000
- - - ◁	19965	256	15.2	OFF	0	0	TEST CELL CALIB
▷	[illegible]	[illegible]	[illegible]	OFF	0	0	TEST CELL CALIB
△	[illegible]	[illegible]	[illegible]	OFF	0	0	TEST CELL CALIB
▽	20950	256	17.5	OFF	0	0	TEST CELL CALIB
◻	19978	4050	25.8	OFF	0	8	LEVEL
[illegible]	19978	4590	25.5	OFF	0	8	LEVEL
+	19978	5260	20.7	OFF	0	9	LEVEL
X	19978	4820	24.0	OFF	0	13	LEVEL
◇	19838	4020	16.5	OFF	0	17	HOVER (OGE)
[illegible]	19838	3860	17.4	OFF	0	18	HOVER (IGE)
[illegible]	19878	5350	24.0	OFF	0	30	LEVEL
[illegible]	20018	2630	22.4	OFF	0	31	LEVEL
[illegible]	19939	5360	9.5	OFF	0	41	LEVEL



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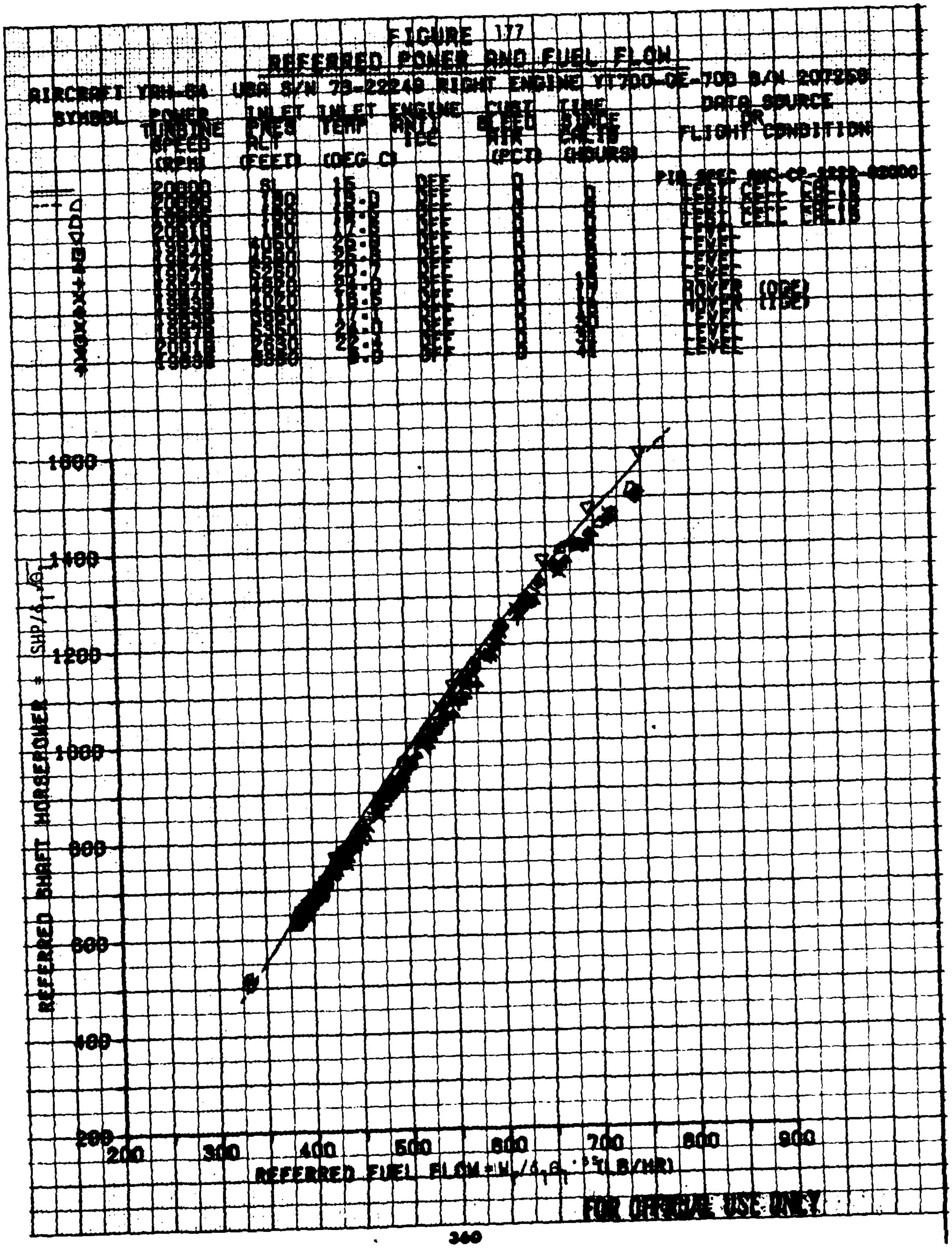
FIGURE 177
REFERRED POWER AND FUEL FLOW
AIRCRAFT YAH-64 USA S/N 73-22249 RIGHT ENGINE YT700-GE-700 S/N 207250
SYMBOL
POWER TURBINE SPEED (RPM)
INLET PRES ALT (FEET)
INLET TEMP (DEG C)
ENGINE ANTI ICE
CUST BLEED AIR (PCT)
TIME SINCE CALIB (HOURS)
DATA SOURCE OR FLIGHT CONDITION
TEST CELL CALIB
LEVEL
HOVER (OGE)
HOVER (IGE)
REFERRED SHAFT HORSEPOWER = SHP/δ√θ
REFERRED FUEL FLOW = W/δθ (LB/HR)
200
400
600
800
1000
1200
1400
1600
200
300
400
500
600
700
800
900
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FIGURE 178

REFERRED POWER AND SPECIFIC FUEL CONSUMPTION

AIRCRAFT YAH-64 USA S/N 73-22249 LEFT ENGINE YT700-GE-700 S/N 207261

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20000	SL	15	OFF	0		PID SPEC AMC-CP-2222-02000
---△	19965	256	16.2	OFF	0	0	TEST CELL CALIB
◇	19990	256	17.0	OFF	0	0	TEST CELL CALIB
▽	20950	256	17.5	OFF	0	0	TEST CELL CALIB
⊡	19979	4050	22.8	OFF	0	8	LEVEL
▲	19979	4590	25.6	OFF	0	8	LEVEL
+	19979	5260	20.7	OFF	0	9	LEVEL
X	19979	4820	24.0	OFF	0	13	LEVEL
◇	19839	4020	16.5	OFF	0	17	HOVER (OGE)
⊠	19839	3860	17.4	OFF	0	18	HOVER (IGE)
⊙	19979	5350	24.0	OFF	0	30	LEVEL
⊠	20019	2680	22.4	OFF	0	31	LEVEL
◆	19839	8360	9.0	OFF	0	41	LEVEL

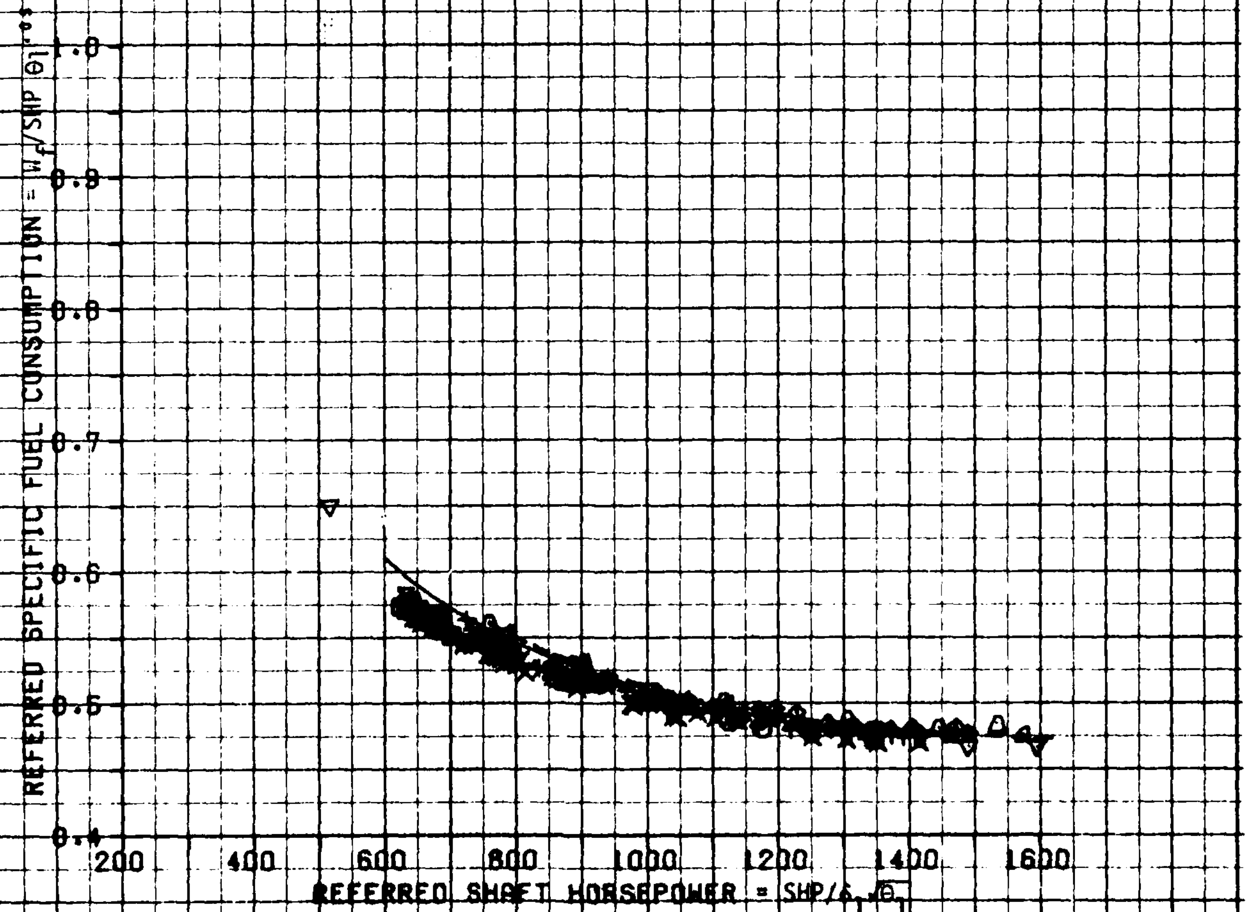

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FIGURE 179

REFERRED POWER AND SPECIFIC FUEL CONSUMPTION

AIRCRAFT YAH-64 USA S/N 73-22249 RIGHT ENGINE YT700-GE-700 S/N 207258

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
	20000	SL	15	OFF	0		PIB SPEC AMC-CP-2222-02000
	20080	180	15.0	OFF	0	0	TEST CELL CALIB
	[illegible]	180	[illegible]	OFF	0	0	TEST CELL CALIB
	20910	180	17.5	OFF	0	0	TEST CELL CALIB
	[illegible]	4060	25.3	OFF	0	[illegible]	LEVEL
	[illegible]	4580	[illegible]	OFF	0	[illegible]	LEVEL
	[illegible]	5260	20.7	OFF	0	[illegible]	LEVEL
	[illegible]	5620	24.0	OFF	0	19	HOVER (OGE)
	[illegible]	7020	15.5	OFF	0	17	HOVER (IGE)
	19838	4020	16.5	OFF	0	18	LEVEL
	19838	3960	17.1	OFF	0	40	LEVEL
	19878	5360	24.0	OFF	0	31	LEVEL
	20018	2630	22.3	OFF	0	41	LEVEL
	19838	5360	5.0	OFF			

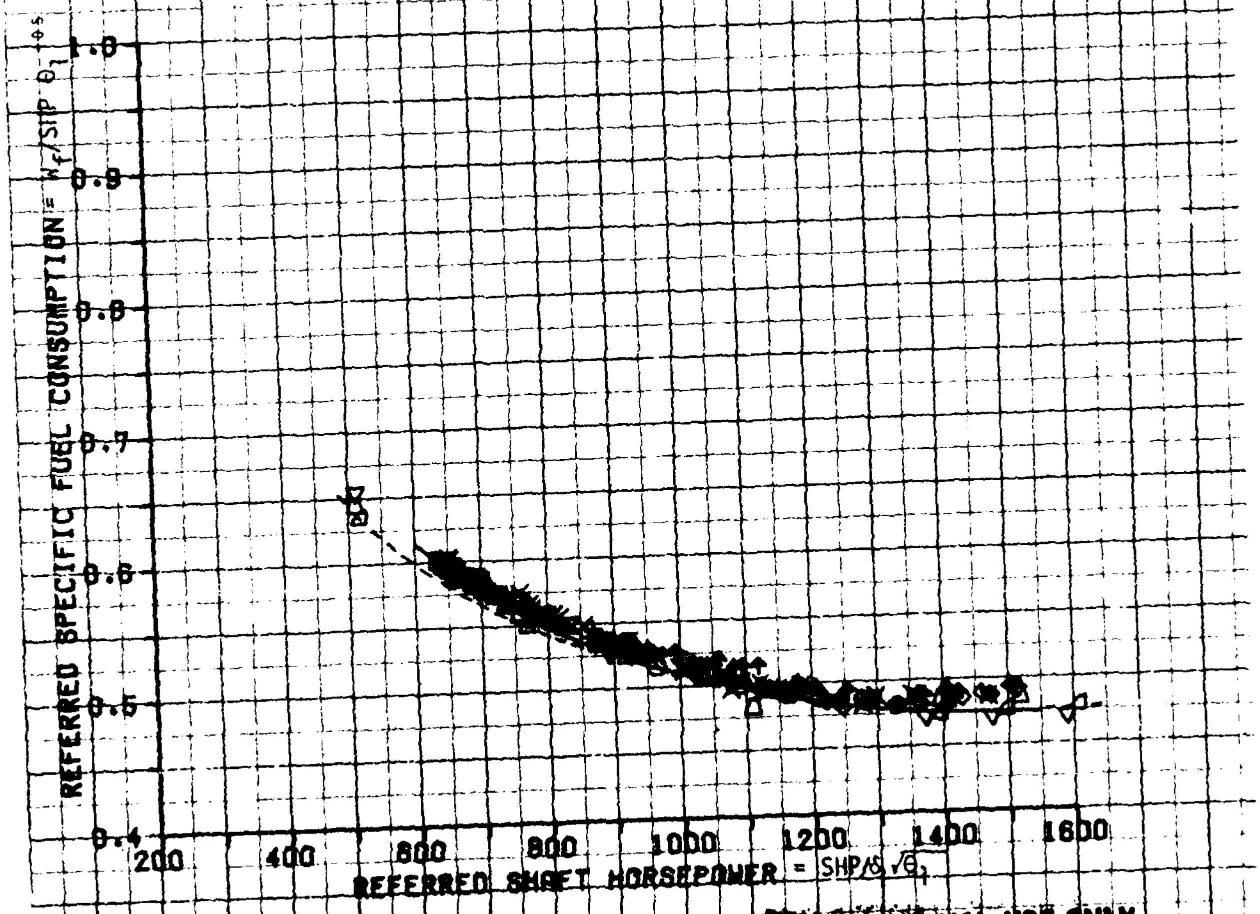

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FIGURE 1[illegible]

REFERRED POWER AND GAS PRODUCER SPEED

AIRCRAFT YAH-64 USA S/N 73-22249 LEFT ENGINE YT700-GE-700 S/N 20728[illegible]

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
——	20000	SL	15	OFF	0		MIL SPEC AMC-CP-2222-02000
---◁	19955	250	15.2	OFF	0	0	TEST CELL CALIB
△	[illegible]	[illegible]	17.0	OFF	0	0	TEST CELL CALIB
▽	20050	250	17.5	OFF	0	0	TEST CELL CALIB
◻	19878	4050	22.8	OFF	0	8	LEVEL
	[illegible]	4500	[illegible]	OFF	0	8	LEVEL
	19879	5260	20.7	OFF	0	8	LEVEL
+	[illegible]	[illegible]	[illegible]	OFF	0	8	LEVEL
X	19878	[illegible]	[illegible]	OFF	0	[illegible]	[illegible]
◇	19839	4020	16.5	OFF	0	17	HOVER (OGE)
⊗	19839	9960	17.[illegible]	OFF	0	18	HOVER (IGE)
	19879	5350	24.0	OFF	0	30	LEVEL
	20019	2680	22.[illegible]	OFF	0	31	LEVEL
✦	19839	9300	9.0	OFF	0	41	LEVEL

NOTE: GE TERMINOLOGY FOR "GAS PRODUCER" IS "GAS GENERATOR."

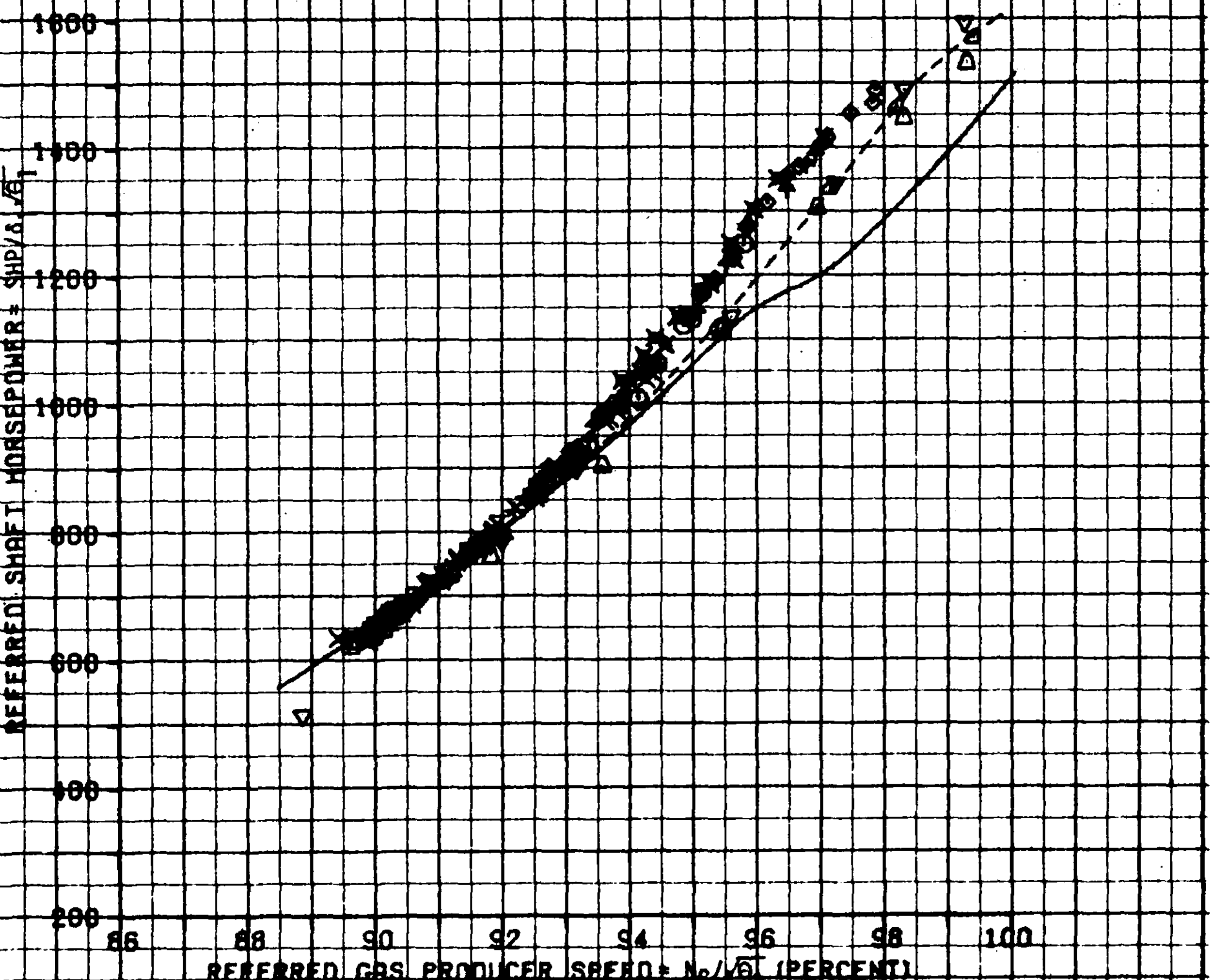

FIGURE 181

REFERRED POWER AND GAS PRODUCER SPEED

AIRCRAFT YAH-64 USA S/N 78-22249 RIGHT ENGINE YT700-GE-700 S/N 207258

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
——	20000	SL	15	OFF	0		PID SPEC [illegible]
---△	20000	180	13.0	OFF	0	0	TEST CELL CALIB
[illegible]	[illegible]	180	[illegible]	OFF	0	0	TEST CELL CALIB
▽	20910	180	[illegible]	OFF	0	0	TEST CELL CALIB
⊡	[illegible]	[illegible]	[illegible]	OFF	0	[illegible]	LEVEL
▲	[illegible]	[illegible]	[illegible]	OFF	0	[illegible]	LEVEL
+	[illegible]	6250	20.7	OFF	0	[illegible]	LEVEL
X	[illegible]	[illegible]	[illegible]	OFF	0	[illegible]	LEVEL
◆	[illegible]	4020	[illegible]	OFF	0	17	HOVER (OGE)
X	[illegible]	[illegible]	[illegible]	OFF	0	[illegible]	HOVER (IGE)
⊙	[illegible]	[illegible]	[illegible]	OFF	0	30	LEVEL
X	20010	2680	22.4	OFF	0	31	LEVEL
✦	[illegible]	[illegible]	[illegible]	OFF	0	[illegible]	LEVEL

NOTE: GE TERMINOLOGY FOR "GAS PRODUCER" IS "GAS GENERATOR."

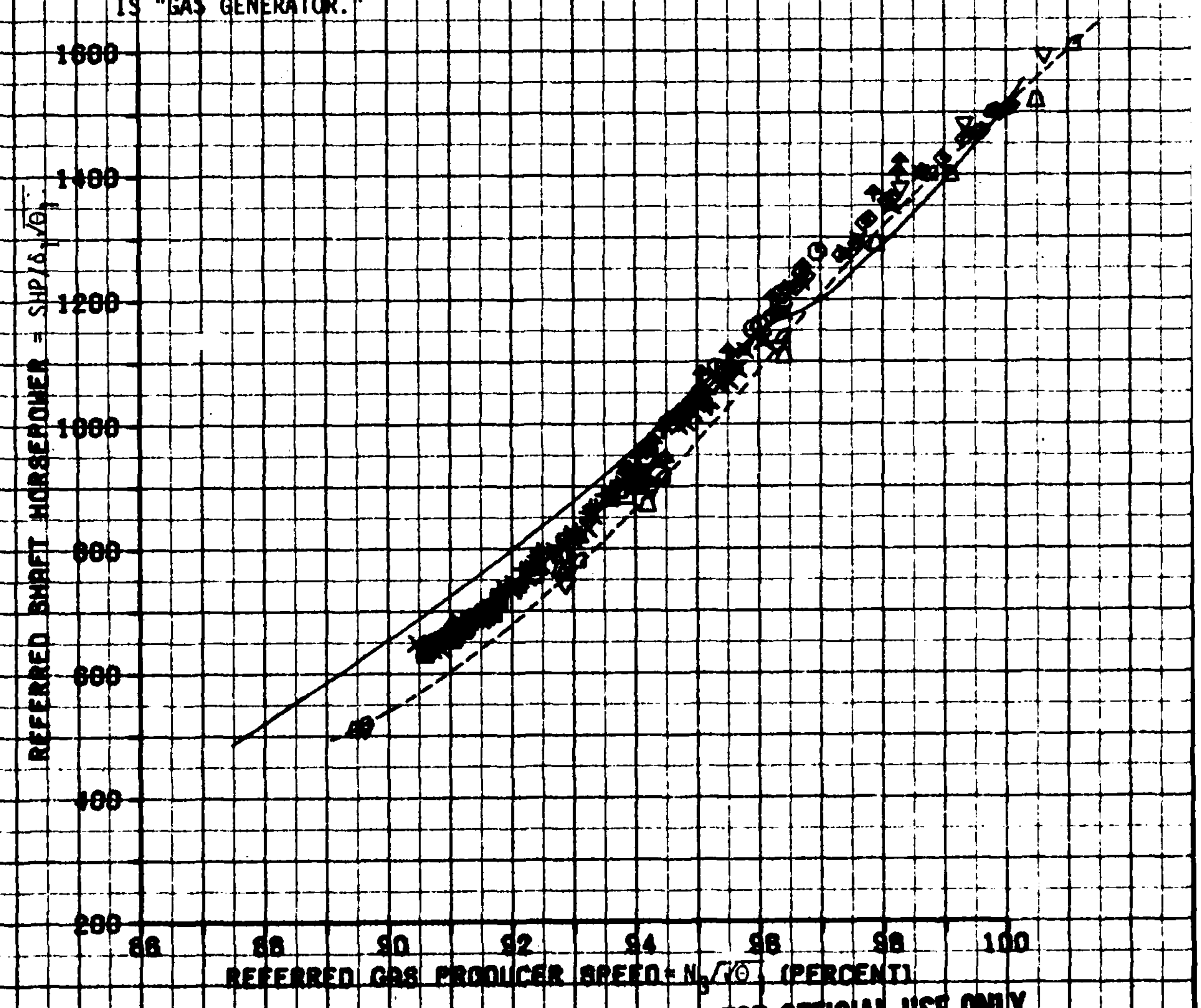

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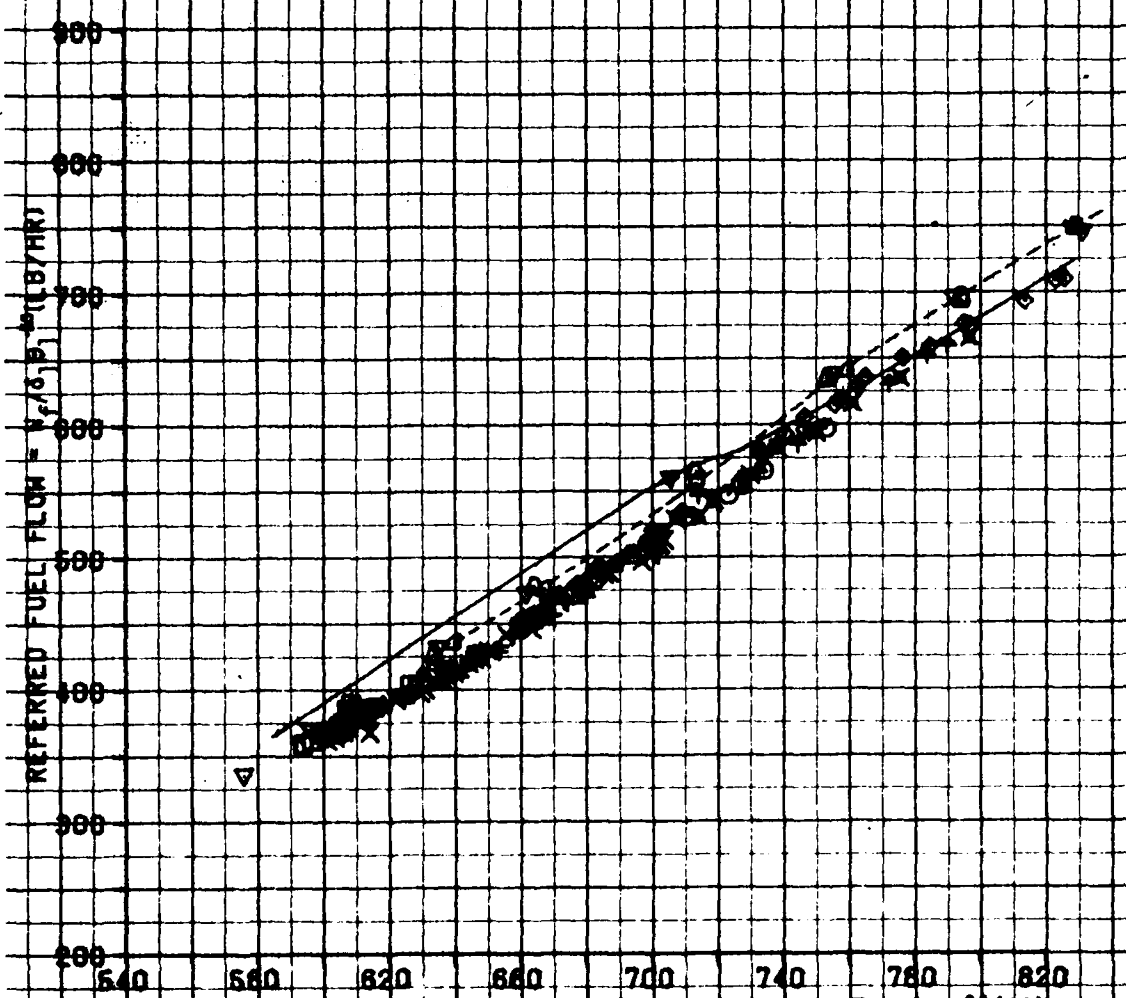
FIGURE 182

REFERRED FUEL FLOW AND GAS TEMPERATURE

AIRCRAFT YAH-64 USA S/N 78-22249 LEFT ENGINE YT700-GE-700 S/N 207281

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
	20000	SL	15	OFF	0		PID SPEC AMC-CP-2222-02000
	19865	256	15.2	OFF	0	0	TEST CELL CALIB
	[illegible]	256?	17.0?	OFF	0	0	TEST CELL CALIB
	20850	256	17.5	OFF	0	0	TEST CELL CALIB
	19979	8050	20.8	OFF	0	8	LEVEL
	19829	4590	25.6	OFF	0	8	LEVEL
	19979	6260	20.7	OFF	0	8	LEVEL
	[illegible]	[illegible]	[illegible]	OFF	0	1?	LEVEL
	19839	4020	16.5	OFF	0	17	HOVER (OGE)
	19839	3960	17.1	OFF	0	18	HOVER (IGE)
	19879	6350	21.0	OFF	0	30	LEVEL
	20019	2680	22.4	OFF	0	35	LEVEL
	19839	6360	9.0	OFF	0	41	LEVEL



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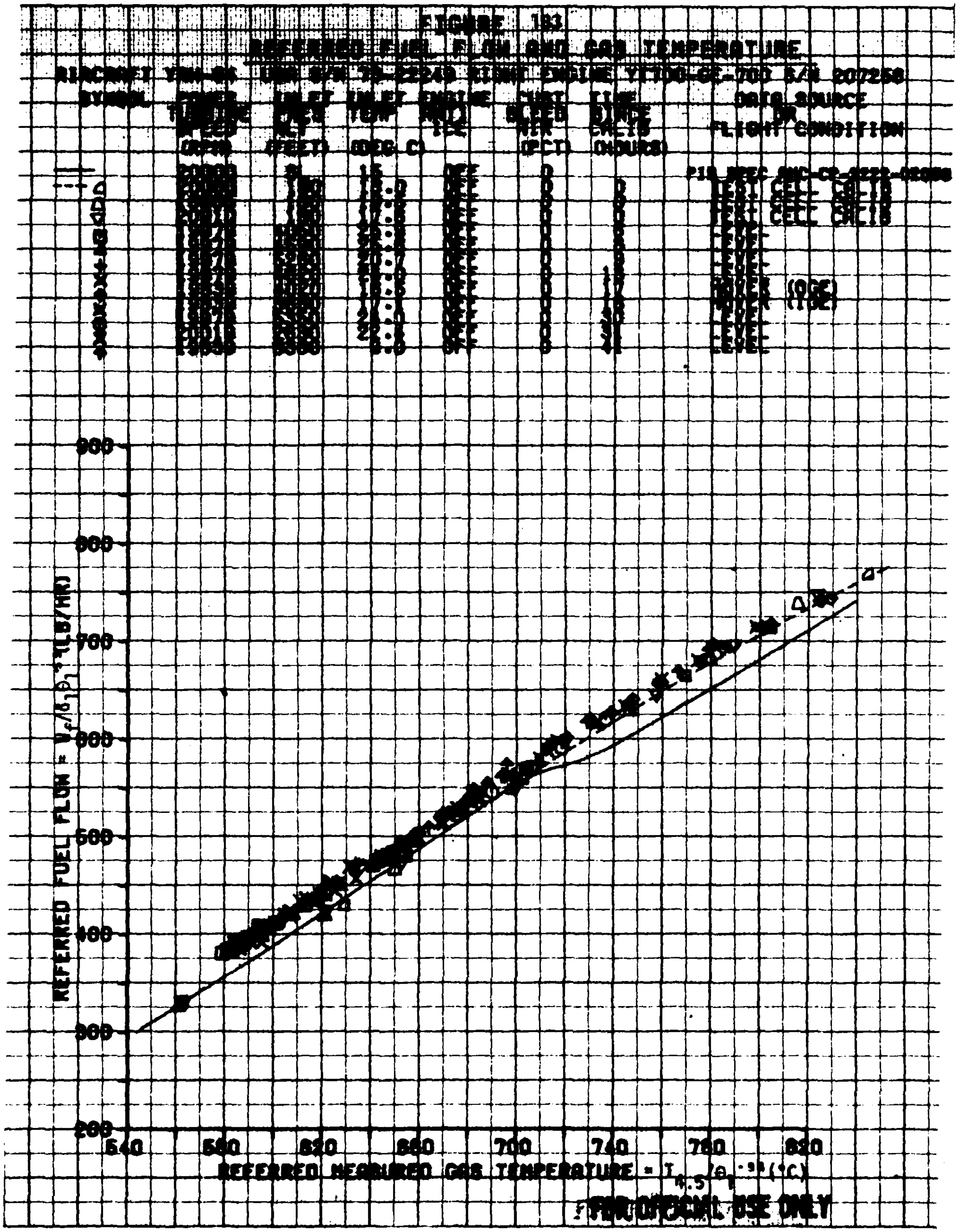
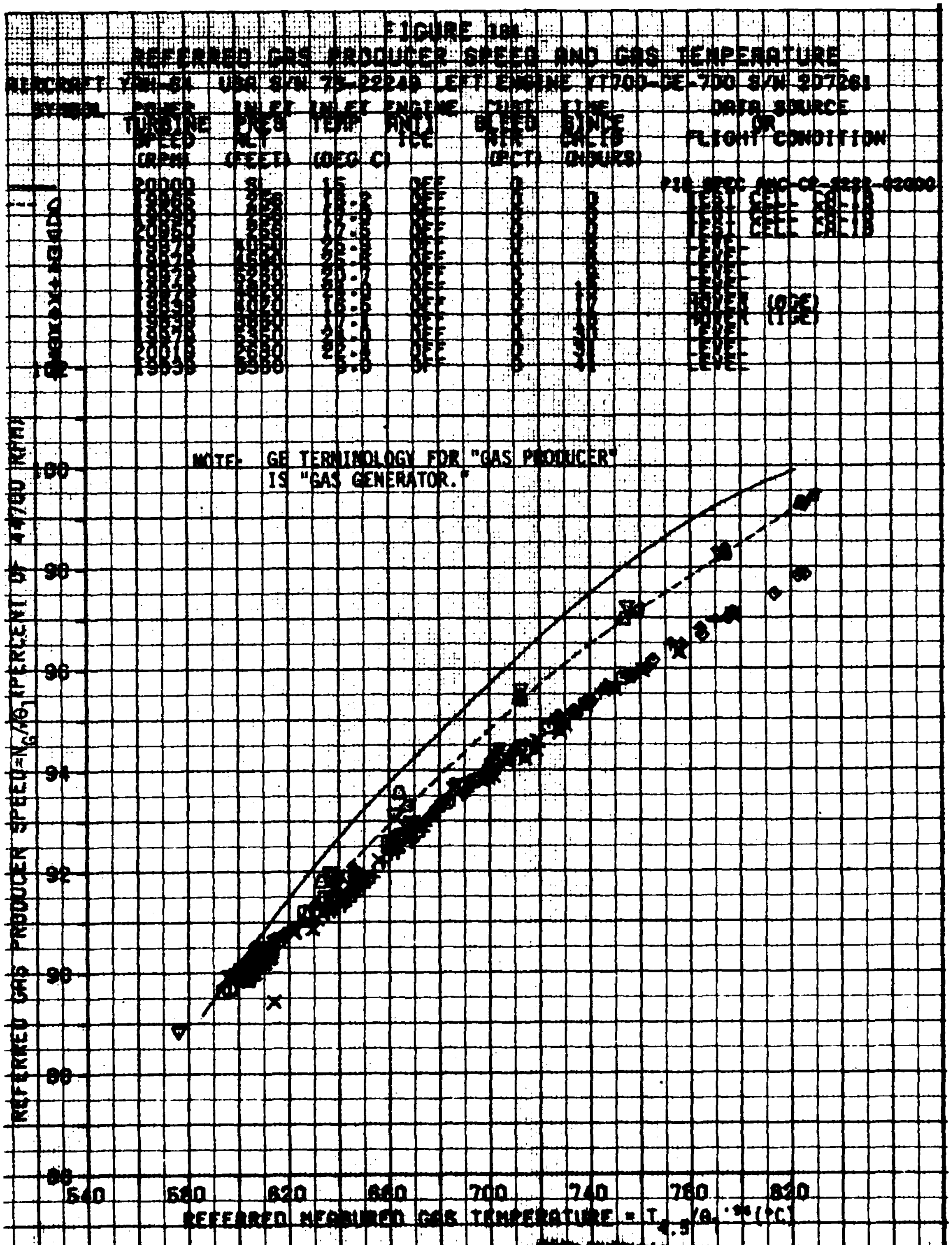
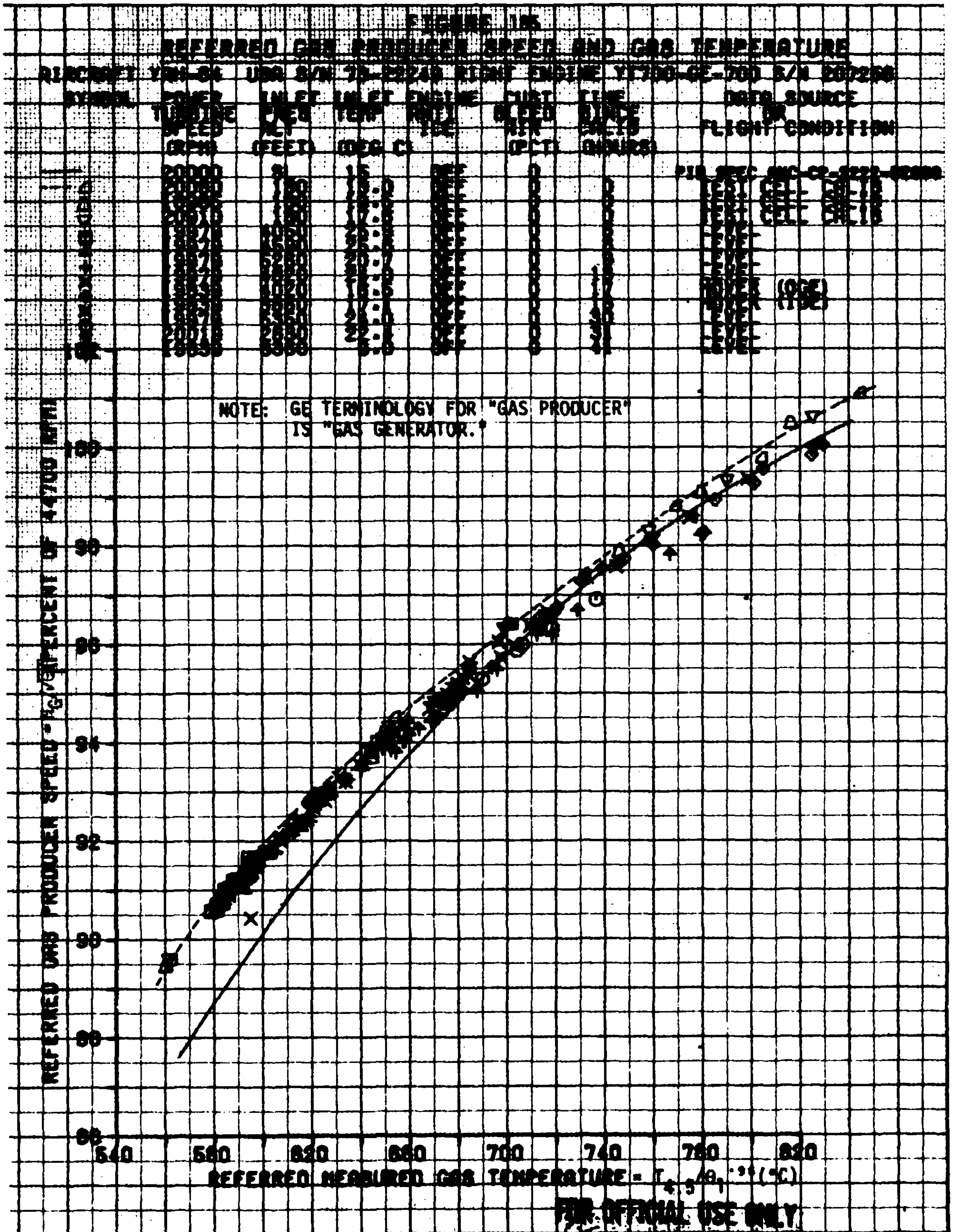

DATA SOURCE
OR
FLIGHT CONDITION
TEST CELL CALIB
LEVEL
HOVER (OGE)
900
800
700
600
500
400
300
200
540
580
620
660
700
740
780
820
REFERRED FUEL FLOW
REFERRED MEASURED GAS TEMPERATURE
FOR OFFICIAL USE ONLY



FOR OFFICIAL USE ONLY



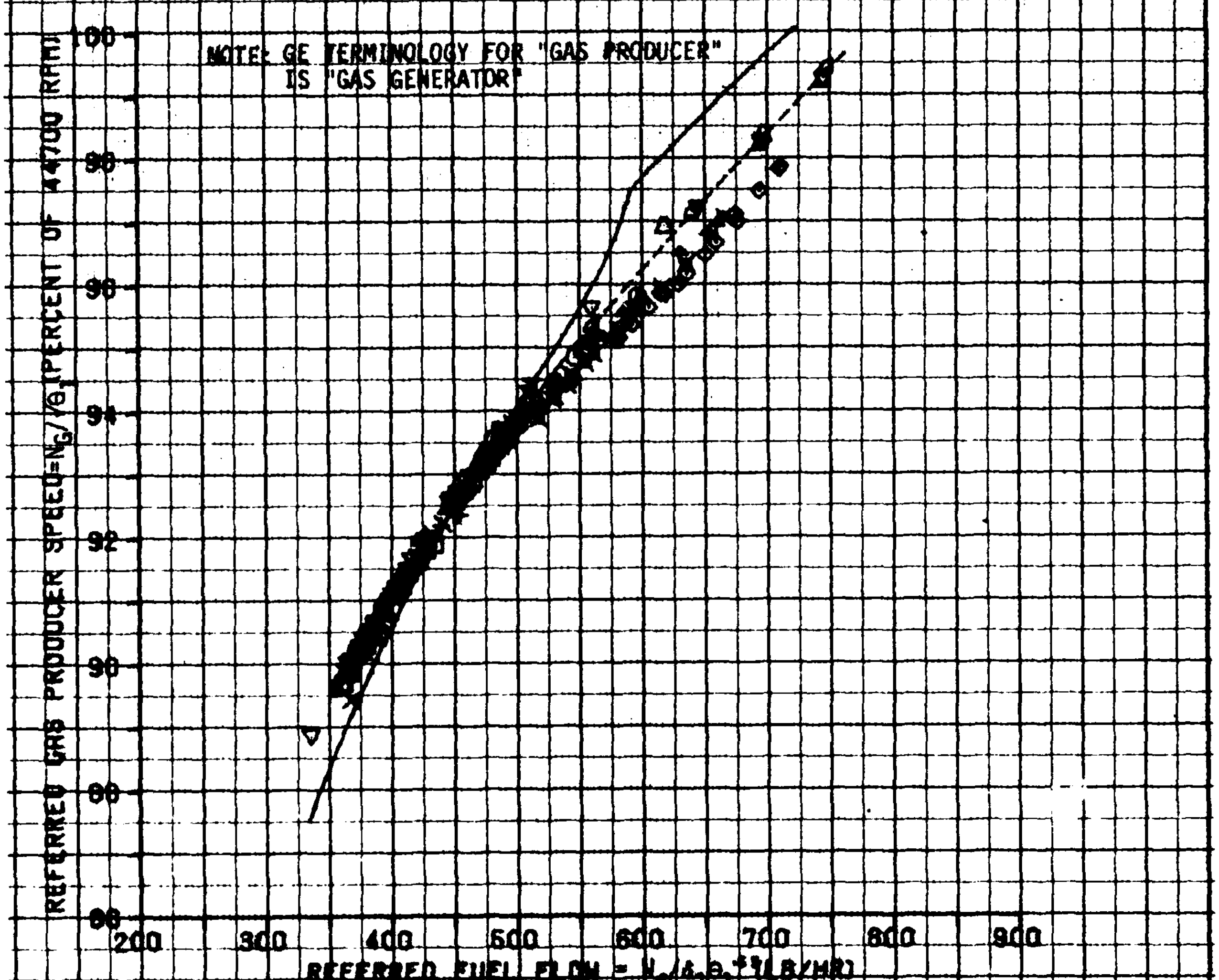
REFERRED GAS PRODUCER SPEED AND GAS TEMPERATURE
DATA SOURCE OR FLIGHT CONDITION
TEST CELL CALIB
LEVEL
HOVER (OGE)
HOVER (IGE)
NOTE: GE TERMINOLOGY FOR "GAS PRODUCER" IS "GAS GENERATOR."
REFERRED MEASURED GAS TEMPERATURE

FIGURE 186

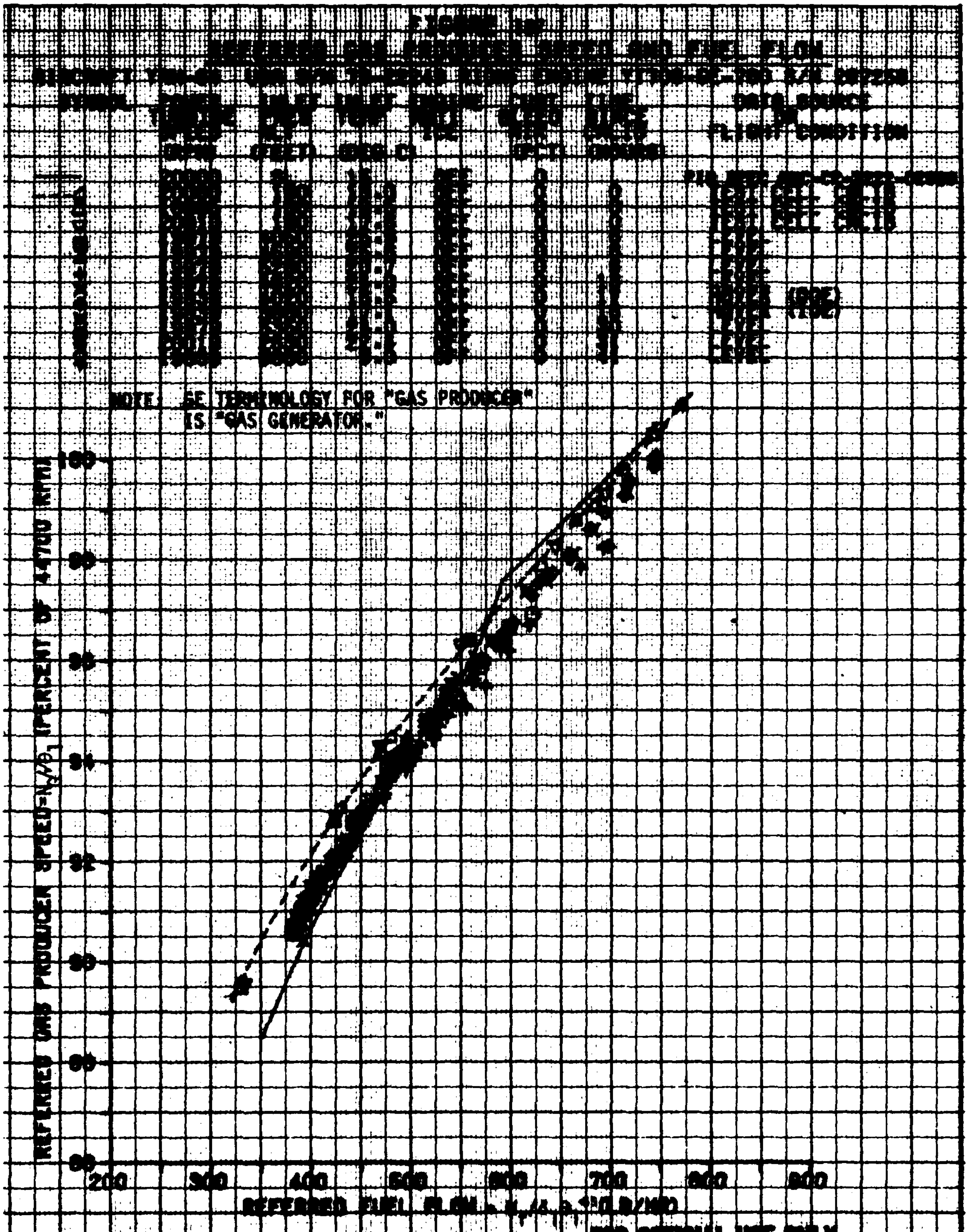
REFERRED GAS PRODUCER SPEED AND FUEL FLOW

AIRCRAFT YAH-64 USA S/N 73-22249 LEFT ENGINE YT700-GE-700 S/N 207261

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (DEG C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
——	20000	SL	15	OFF	0		PID SPEC AMC-CP-2222-02000
- - - △	19955	256	16.2	OFF	0	0	TEST CELL CALIB
[illegible]	19950	256	17.8	OFF	0	0	TEST CELL CALIB
[illegible]	20950	256	17.5	OFF	0	0	TEST CELL CALIB
[illegible]	19979	4050	25.9	OFF	0	8	LEVEL
[illegible]	19879	4550	25.8	OFF	0	8	LEVEL
[illegible]	19979	5260	20.7	OFF	0	8	LEVEL
[illegible]	[illegible]	[illegible]	21.0	OFF	0	17	LEVEL
[illegible]	19839	6020	16.5	OFF	0	17	HOVER (OGE)
[illegible]	19839	3950	17.1	OFF	0	18	HOVER (IGE)
[illegible]	19879	6350	24.0	OFF	0	30	LEVEL
[illegible]	20019	2680	22.4	OFF	0	31	LEVEL
[illegible]	19839	6380	9.0	OFF	0	41	LEVEL



FOR OFFICIAL USE ONLY



NOTE: GE TERMINOLOGY FOR "GAS PRODUCER"
IS "GAS GENERATOR."
100
200
300
400
500
600
700
800
900
REFERRED FUEL FLOW

The following figures are omitted due to CONFIDENTIAL classification:

Figure 188. Referred Power and Compressor Discharge Pressure. (U)

Figure 189. Referred Power and Compressor Discharge Pressure. (U)

Figure 190. Referred Compressor Discharge Pressure and Gas Temperature. (U)

Figure 191. Referred Compressor Discharge Pressure and Gas Temperature. (U)

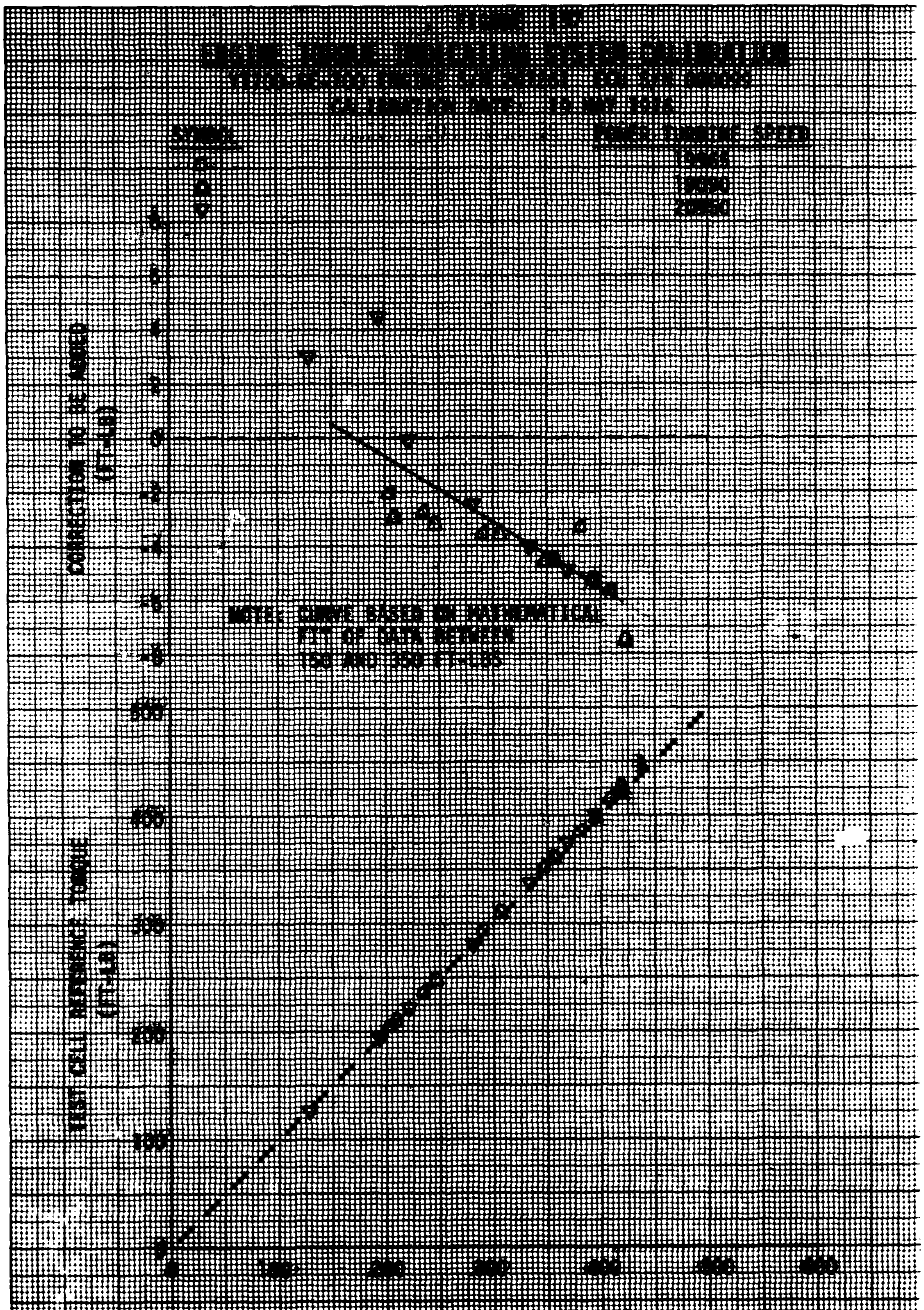
Figure 192. Referred Compressor Discharge Pressure and Fuel Flow. (U)

Figure 193. Referred Compressor Discharge Pressure and Fuel Flow. (U)

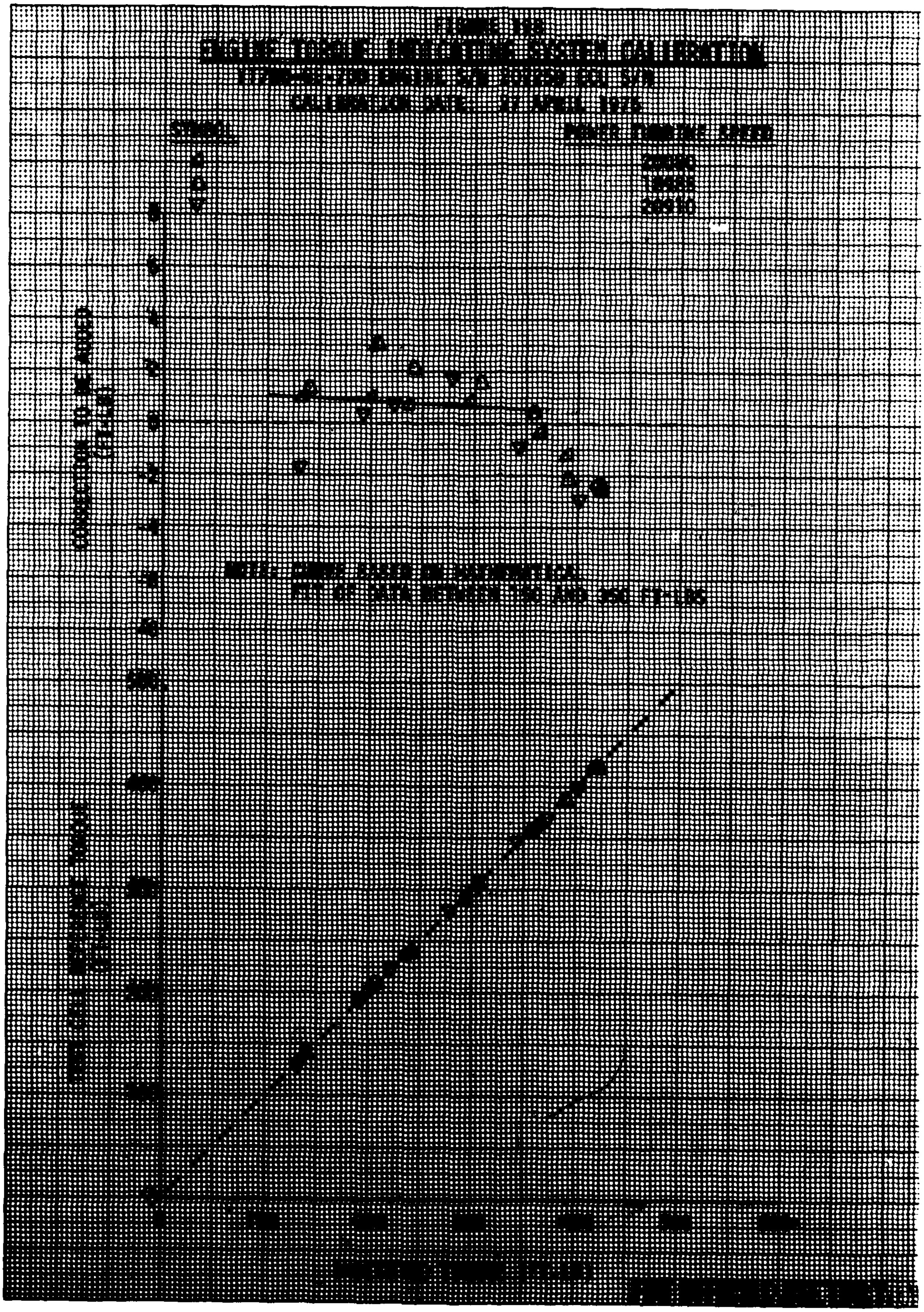
Figure 194. Referred Compressor Discharge Pressure and Gas Producer Speed. (U)

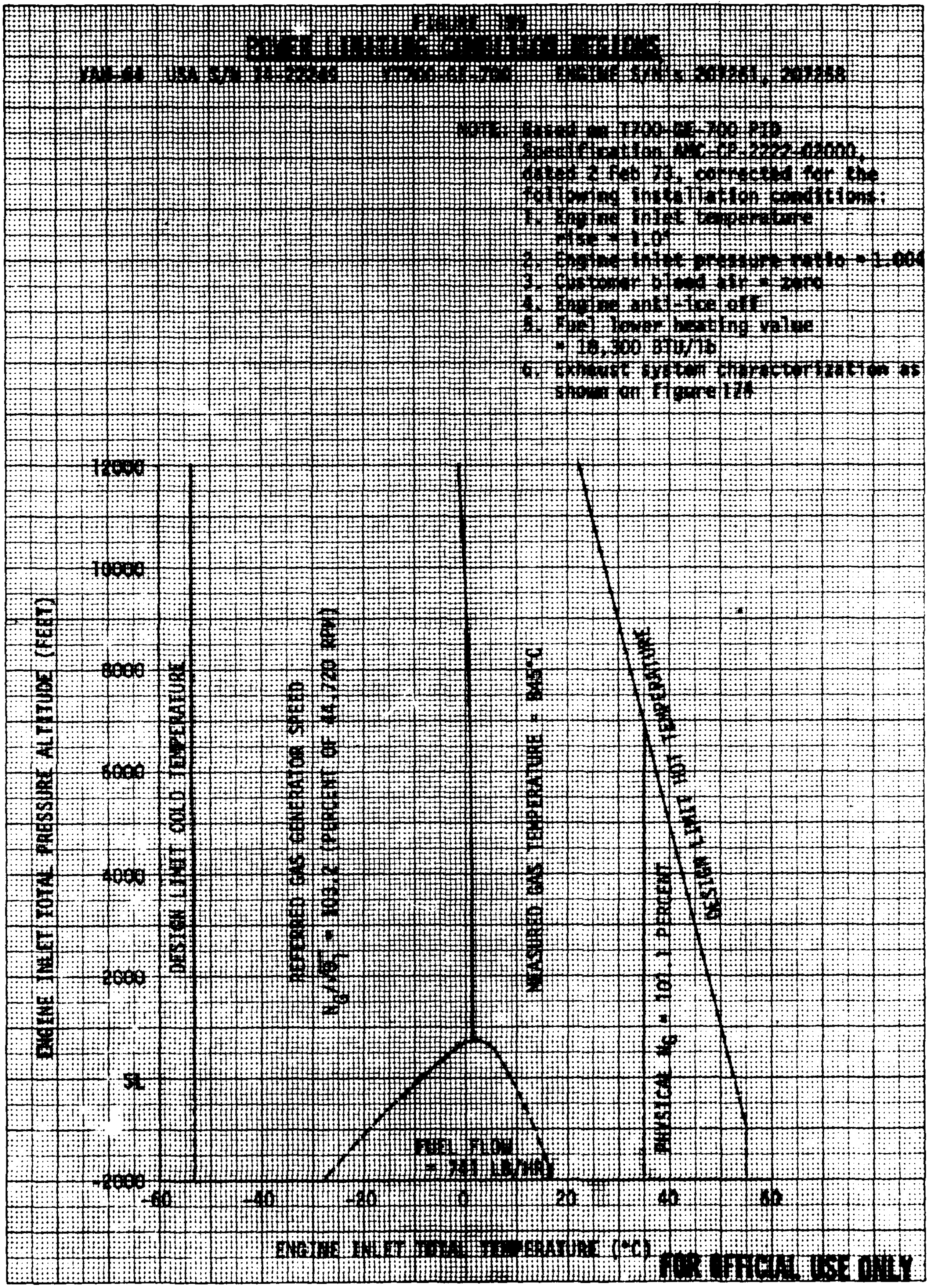
Figure 195. Referred Compressor Discharge Pressure and Gas Producer Speed. (U)

Figure 196. Engine Operating Schedule. (U)

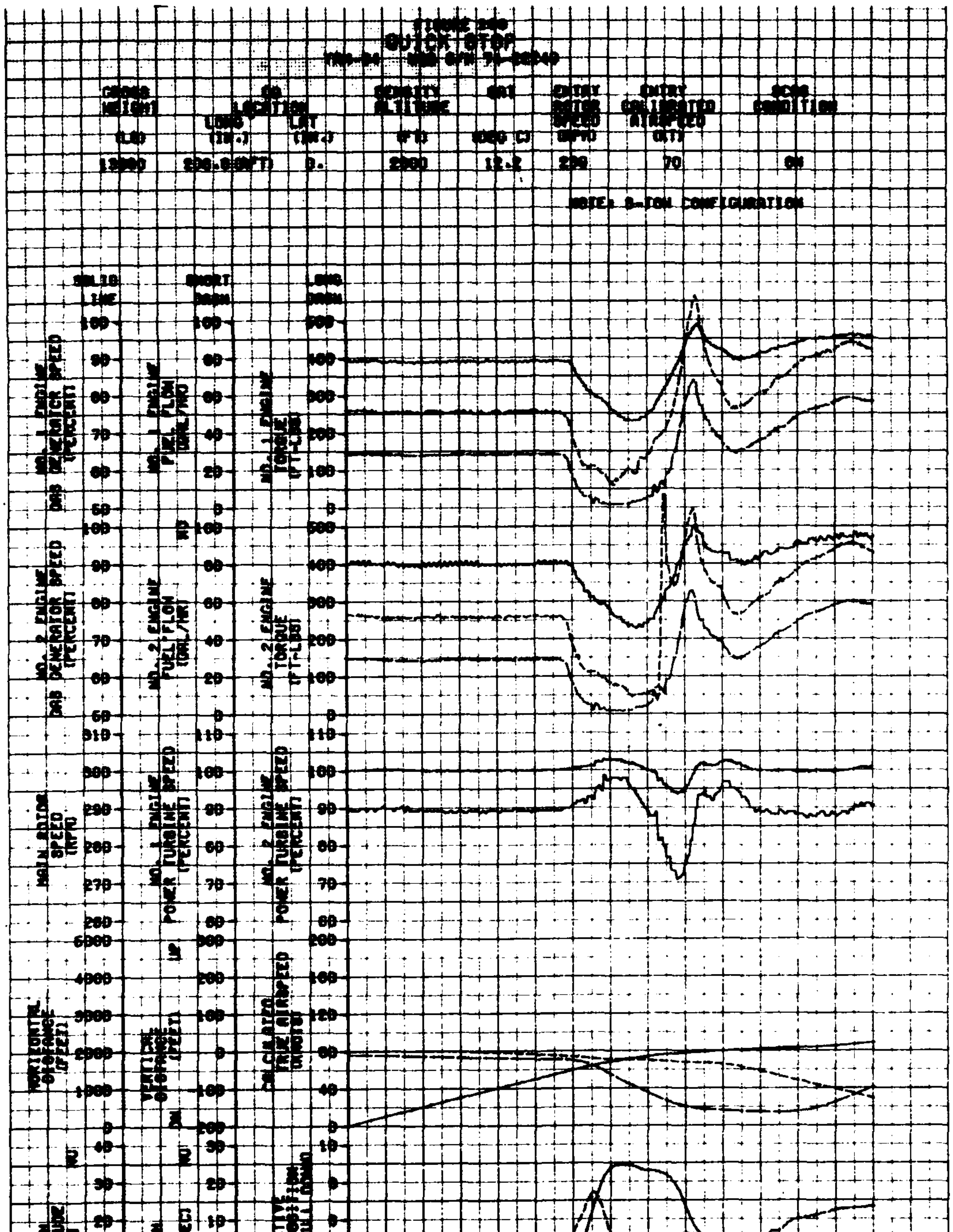


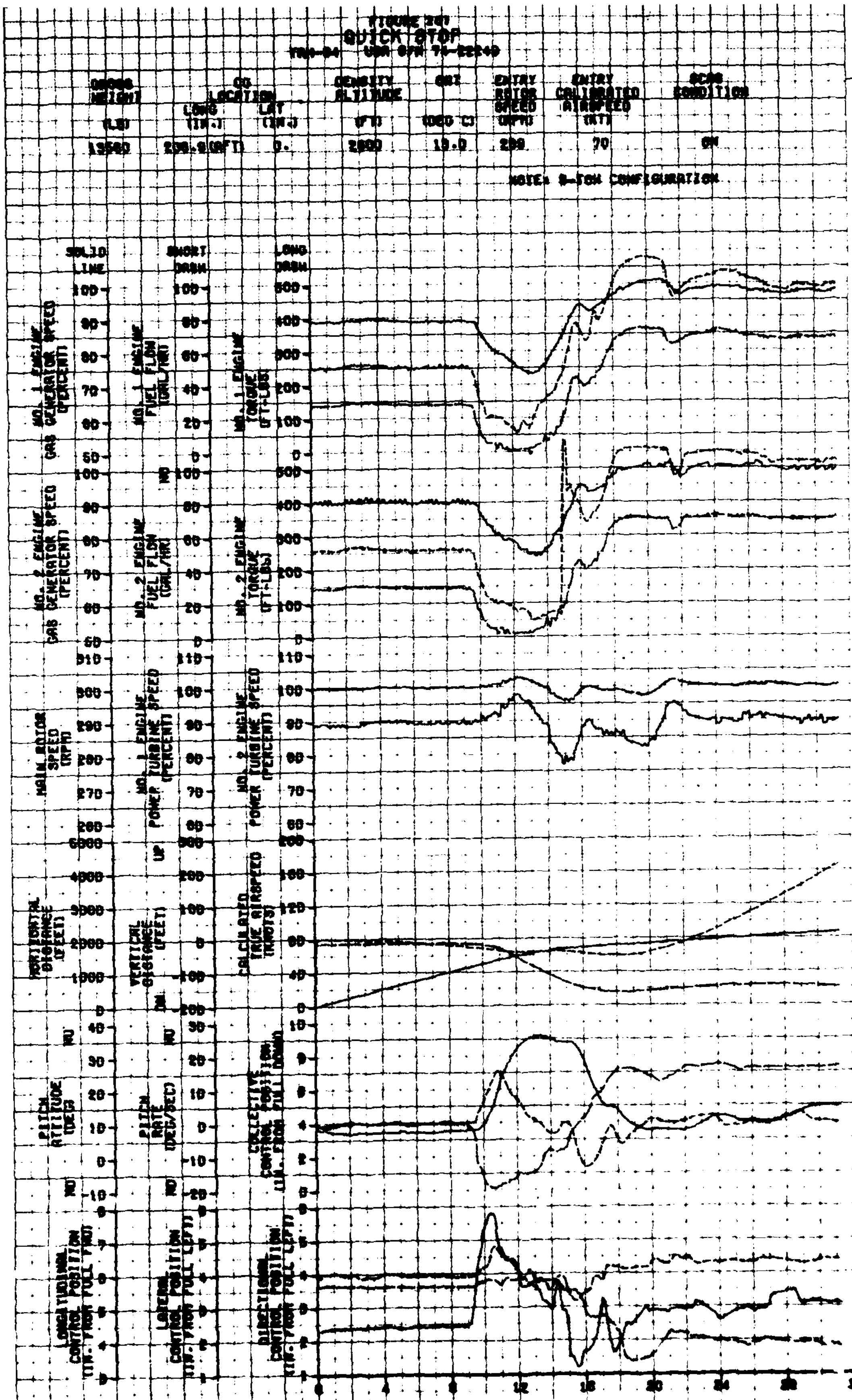
FOR OFFICIAL USE ONLY

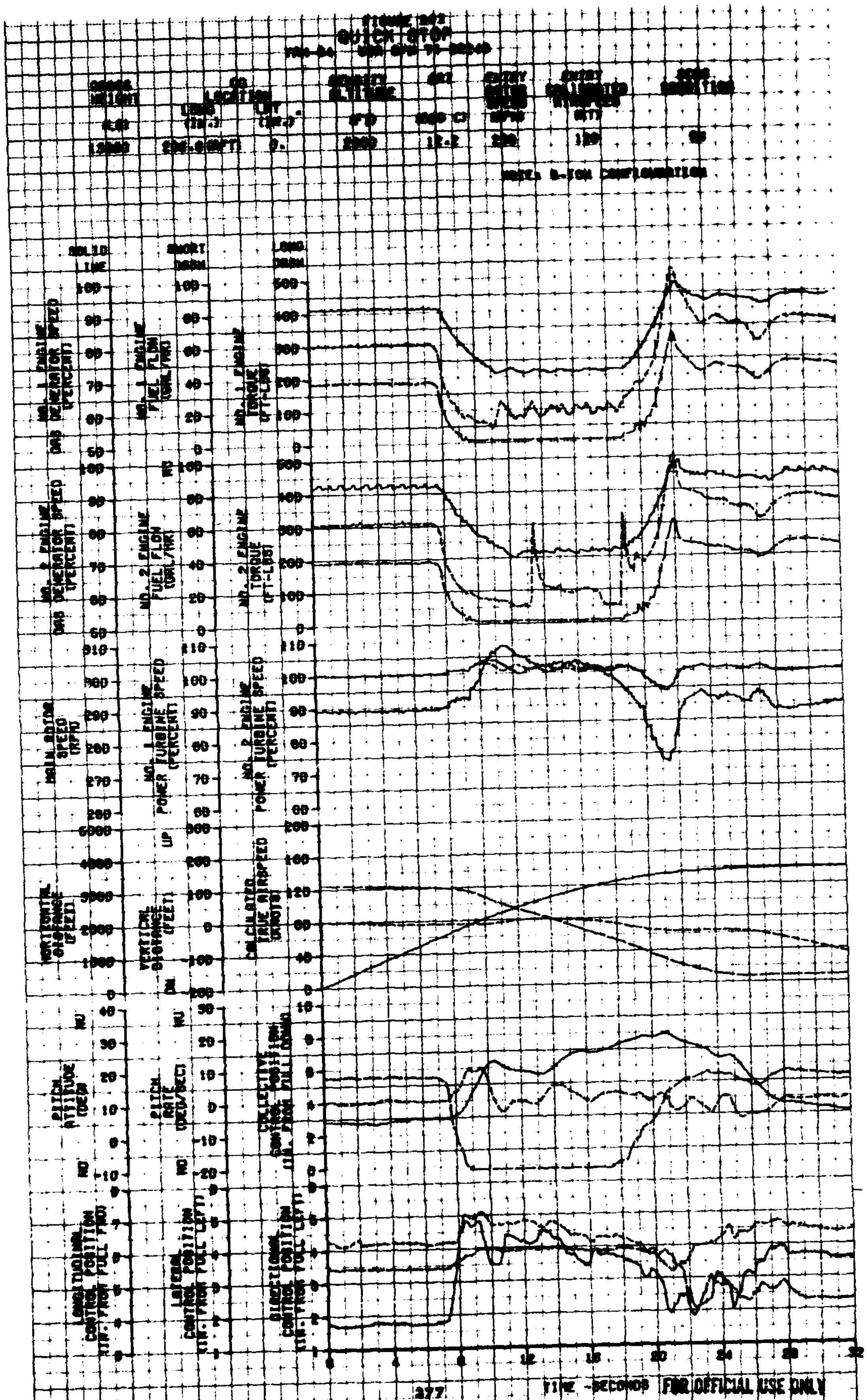

ENGINE TORQUE INDICATING SYSTEM CALIBRATION
CALIBRATION DATE:
SYMBOL
POWER TURBINE SPEED



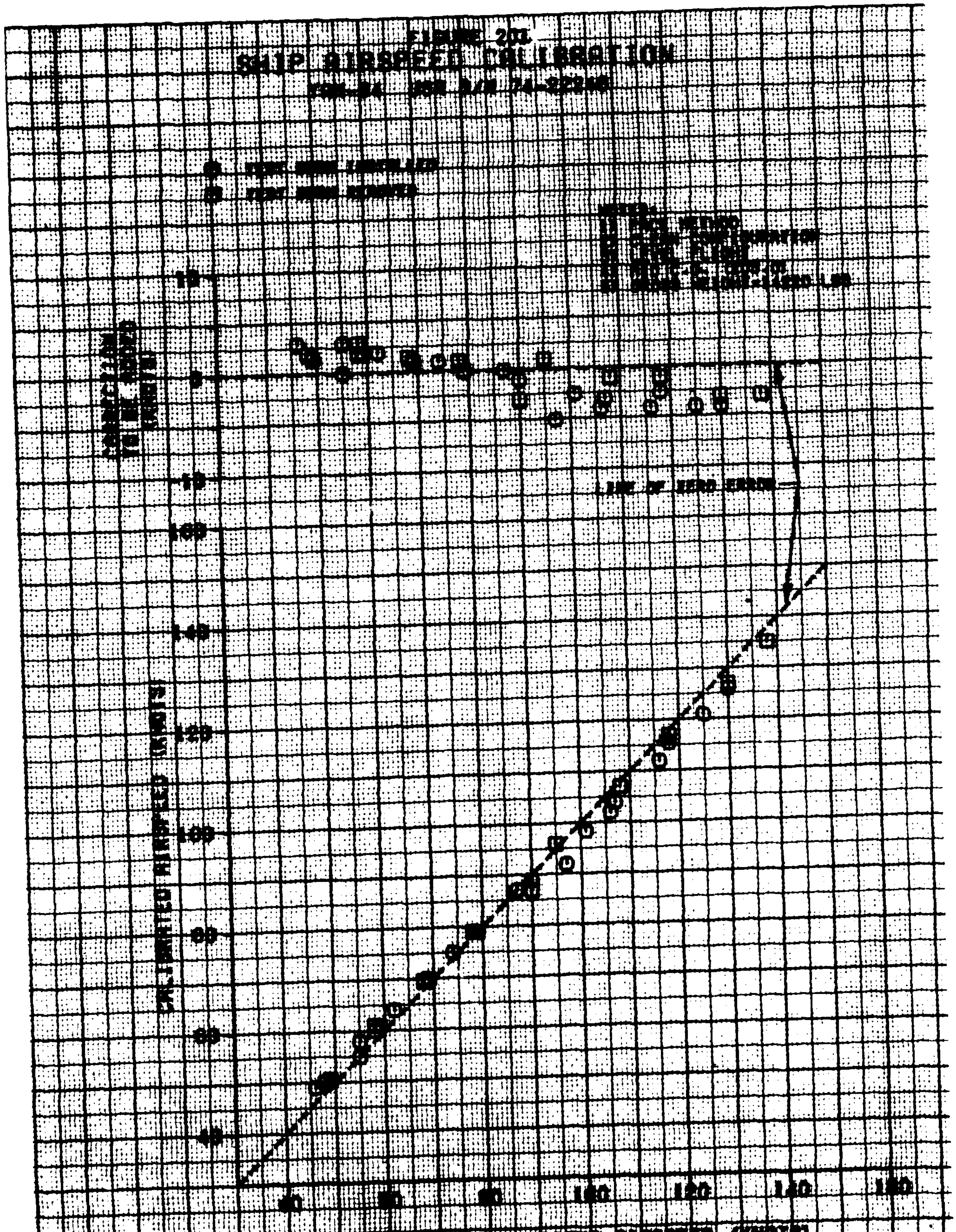
POWER LIMITING CONDITION REGIONS
NOTE: Based on T700-GE-700 PID
Specification AMC-CP-2222-02000,
dated 2 Feb 73, corrected for the
following installation conditions:
1. Engine inlet temperature
rise = 1.0°
2. Engine inlet pressure ratio = 1.000
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value
= 18,300 BTU/lb
6. Exhaust system characterization as
shown on Figure 178
12000
10000
8000
6000
4000
2000
SL
-2000
-60
-40
-20
0
20
40
60
ENGINE INLET TOTAL PRESSURE ALTITUDE (FEET)
ENGINE INLET TOTAL TEMPERATURE (°C)
DESIGN LIMIT COLD TEMPERATURE
REFERRED GAS GENERATOR SPEED
MEASURED GAS TEMPERATURE
PHYSICAL N_G = 101.1 PERCENT
DESIGN LIMIT HOT TEMPERATURE
FUEL FLOW


QUICK STOP
GROSS WEIGHT
LOCATION
LONG
LAT
DENSITY ALTITUDE
ENTRY ROTOR SPEED
ENTRY CALIBRATED AIRSPEED
CONDITION
13000
0.
2000
12.2
70
NOTE: 8-TON CONFIGURATION
SOLID LINE
SHORT DASH
LONG DASH
NO. 1 ENGINE GAS GENERATOR SPEED (PERCENT)
NO. 1 ENGINE FUEL FLOW
NO. 1 ENGINE TORQUE
NO. 2 ENGINE GAS GENERATOR SPEED (PERCENT)
NO. 2 ENGINE FUEL FLOW (GAL/HR)
NO. 2 ENGINE TORQUE
MAIN ROTOR SPEED (RPM)
NO. 1 ENGINE POWER TURBINE SPEED (PERCENT)
NO. 2 ENGINE POWER TURBINE SPEED (PERCENT)
HORIZONTAL DISTANCE (FEET)
VERTICAL DISTANCE (FEET)
CALCULATED TRUE AIRSPEED (KNOTS)
PITCH ATTITUDE (DEG)
PITCH RATE (DEG/SEC)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
0
4
8
12
16
20
24
28
32
TIME - SECONDS


QUICK STOP
GROSS WEIGHT (LB) 13580
CG LOCATION LONG (IN.) LAT (IN.) 0.
DENSITY ALTITUDE (FT) 2800
(DEG C) 19.0
ENTRY ROTOR SPEED (RPM)
ENTRY CALIBRATED AIRSPEED (KT) 70
SCAS CONDITION ON
SOLID LINE
SHORT DASH
LONG DASH
NO. 1 ENGINE GAS GENERATOR SPEED (PERCENT)
NO. 1 ENGINE FUEL FLOW (GAL/HR)
NO. 1 ENGINE TORQUE (FT-LBS)
NO. 2 ENGINE GAS GENERATOR SPEED (PERCENT)
NO. 2 ENGINE FUEL FLOW (GAL/HR)
NO. 2 ENGINE TORQUE (FT-LBS)
MAIN ROTOR SPEED (RPM)
NO. 1 ENGINE POWER TURBINE SPEED (PERCENT)
NO. 2 ENGINE POWER TURBINE SPEED (PERCENT)
HORIZONTAL DISTANCE (FEET)
VERTICAL DISTANCE (FEET)
CALCULATED TRUE AIRSPEED (KNOTS)
PITCH ATTITUDE (DEG)
PITCH RATE (DEG/SEC)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
TIME - SECONDS



QUICK STOP
SOLID LINE
SHORT DASH
LONG DASH
NO. 1 ENGINE FUEL FLOW (GAL/HR)
NO. 1 ENGINE TORQUE (FT-LBS)
NO. 2 ENGINE FUEL FLOW (GAL/HR)
NO. 2 ENGINE TORQUE (FT-LBS)
MAIN ROTOR SPEED (RPM)
NO. 1 ENGINE POWER TURBINE SPEED (PERCENT)
NO. 2 ENGINE POWER TURBINE SPEED (PERCENT)
HORIZONTAL DISTANCE (FEET)
VERTICAL DISTANCE (FEET)
CALCULATED TRUE AIRSPEED (KNOTS)
PITCH ATTITUDE (DEG)
PITCH RATE (DEG/SEC)
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
TIME - SECONDS



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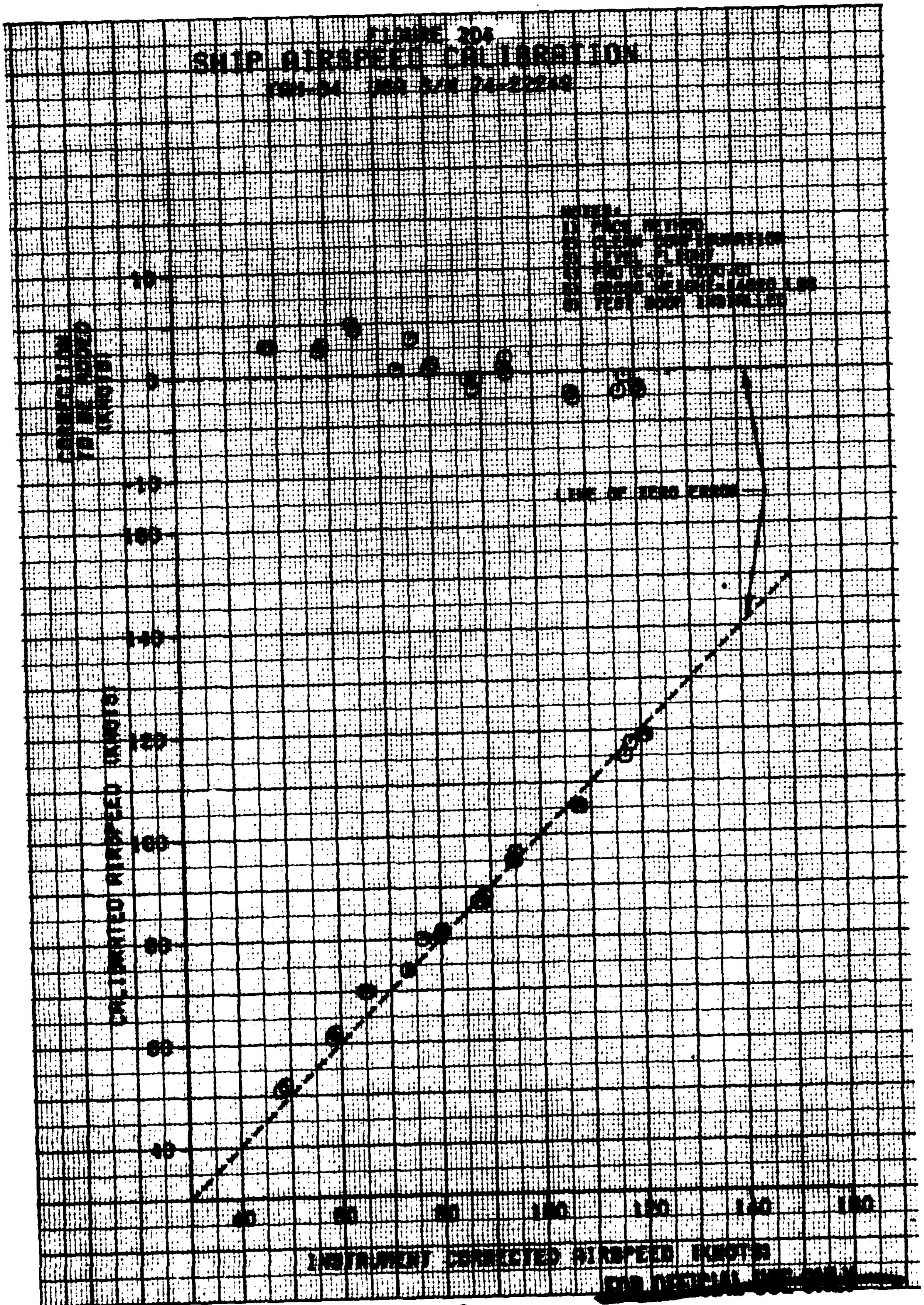

FIGURE 204
SHIP AIRSPEED CALIBRATION
NOTES:
10
0
-10
LINE OF ZERO ERROR
160
140
120
100
80
60
40
CALIBRATED AIRSPEED (KNOTS)
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

APPENDIX H. EQUIPMENT PERFORMANCE REPORTS

EPR Number	Date Submitted	Descriptive Title
74-07-02-01	10 Jul 76	Pneumatic engine starting system failure
-02	19 Jul 76	Malfunction in starter surge valve
-03	14 Jul 76	Requirement to turn ECS off while starting engine
-04	14 Jul 76	ECU lockout malfunction
-05	14 Jul 76	Pedal trim clutch release
-06	13 Jul 76	Force augmentation system malfunction
-07	12 Jul 76	Pitch oscillations from the ASE computer
-08	14 Jul 76	APU abort start
-09	15 Jul 76	Intermittent electrical contact in the rotor brake switch
-10	16 Jul 76	Failure of APU starter solenoid
-11	19 Jul 76	APU starter thermal valve malfunction
-12	17 Jul 76	APU hung start
-13	17 Jul 76	SDC not switching to IPAS after engine start
-14	17 Jul 76	Pitch SAS disengagement in descending left turn
-15	17 Jul 76	Main transmission oil pressure transmitter switch faulty
-16	17 Jul 76	Main transmission low oil pressure transmitter switch faulty

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EPR Number	Date Submitted	Descriptive Title
-17	17 Jul 76	Intermittent electrical connection of #1 fuel switch
-18	17 Jul 76	Grabbing and overheating of right brake assembly
-19	19 Jul 76	UHF radio receiver intermittent
-20	21 Jul 76	Cyclic control jam from FOD
-21	22 Jul 76	Faulty main transmission #1 oil pressure transmitter
-22	22 Jul 76	EMI on UHF from red strobe light
-23	22 Jul 76	Leaking IPAS
-24	23 Jul 76	Precessing RMI
-25	23 Jul 76	Intermittent SAS inputs
-26	30 Jul 76	Faulty main transmission #2 oil pressure transmitter
-27	30 Jul 76	Faulty accessory gearbox oil pressure transmitter
-28	13 Aug 76	IR fan FOD'ed
-29	13 Aug 76	IR fan failure during turn-up
-30	16 Aug 76	VSD malfunction
-31	16 Aug 76	Zero shift in Marconi fuel gauge
-32	16 Aug 76	SAS yaw hardover during flight
-33	18 Aug 76	Marconi gauges intermittently failed
-34	18 Aug 76	VSD failure
-35	19 Aug 76	VSD failure
-36	23 Aug 76	APU starting malfunction
-37	26 Aug 76	Leaking IPAS

[redacted]

EPR Number	Date Submitted	Descriptive Title
-38	29 Aug 76	APU starting malfunction
-39	30 Aug 76	Faulty #2 engine starter
-40	7 Sep 76	APU starting malfunction
-41	17 Sep 76	APU starting malfunction

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